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PERFORMANCE FLIGHT TEST TECHNIQUES

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preface

While many reports, articles, and texts have been written concerning flight testing, most have approached the subject from the viewpoint of the engineer or theorist. Where some mention has been made of the way the aircraft is flown while recording data for a test, the author has generally been non-rated and has limited his comments to general procedure. Faced with the problem of training experimental test pilots, the USAF Aerospace Research Pilot School recognized the need for a handbook written primarily for test pilots. This performance flight test handbook for pilots is the result.

The content of this handbook is not as technical as that required by the flight test engineer but does present the theoretical basis of each performance flight test on the level the average pilot can understand. Enough of the procedure for reducing the flight test data to a usable form is given so that the pilot can understand the engineer's problems in handling the data. In many cases this has been simplified in order to reduce the work load and because of time limitations placed on the student at the School.

The major emphasis of this handbook is placed on the actual flight test techniques or "tricks of the trade" used in flying each performance test so that the pilot new to the field may benefit by the experience of others. The reader should bear in mind that since this handbook was designed specifically for school use, certain procedures as given for the tests are dictated by school conditions. However, the techniques involved in flying the different tests should be applicable anywhere.

It is hoped that this handbook will not only be an aid to the student attending the School but will become a ready reference for him and for others engaged in the flying of performance flight tests.

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introduction

From the viewpoint of the pilot the technique of performance flight testing can be summarized by the three key words: ORGANIZATION, ATTITUDE, and TRIM. The performance test pilot who understands and flies by these three key words will do much to make his job easier and and to improve the quality of the test data he obtains.

ORGANIZATION: To the test pilot this work should mean the proper organization and planning of his flight. He should plan each test flight in detail so that he knows exactly what to do and when. The procedure and technique for the test should be second nature to him as well as the proper operation of any required equipment. With the high cost of flight test time and the limited endurance of some of the later fighter aircraft, it is of utmost importance to make the most of every minute in the air.

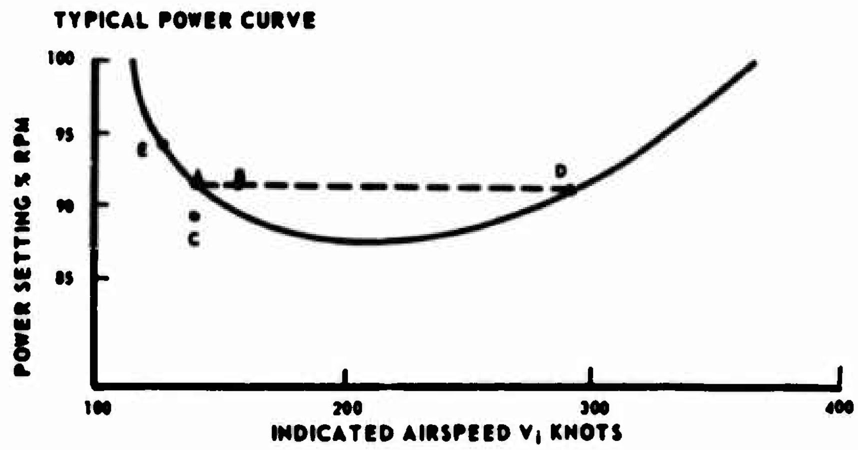
ATTITUDE: Attitude flying is nothing more than using the outside horizon as a flight indicator and using this reference the same way as the instrument pilot uses the artificial horizon in the cockpit. Since this makes quite a large instrument, is not subject to precession errors, and small changes in aircraft attitude can be noted, the precision of one's flying can be greatly improved. The modern pilot, being accustomed to flying on instruments with head in the cockpit IFR or VFR, will probably find it a little difficult at first to adjust to this concept.

In order to use the technique of attitude flying a clearly defined horizon is needed. Sometimes weather conditions will not permit this and the pilot has to settle for a cloud formation as far away as possible.

A prominent point on the canopy or nose of the aircraft is needed as a reference point. This may consist of the general outline of the nose or canopy, a line of rivets, or a grease pencil mark or smudge on the windshield. The pilot then aligns this reference mark on the aircraft with the horizon much the same as using a fixed gun sight. Care should be taken to keep the head in the same place since every time the head is moved the reference will change.

Using this technique and keeping in mind that for every engine power setting and aircraft attitude there is a corresponding airspeed, very small corrections to the speed of the aircraft can be made. Remember that as the nose is raised on the horizon the airspeed decreases and as the nose is lowered on the horizon the airspeed increases. Care should be taken to allow sufficient time for the lag in the aircraft instruments before making further corrections to the attitude of the aircraft.

Two basic techniques are available for maneuvering the aircraft into a stabilized condition, depending upon whether the desired speed range is on the "front side" or "back side" of the power curve.



In the preceding figure is a sample power curve for a typical jet aircraft at medium altitude. The speed range from approximately 230 knots to V_{max} is the front side of the power curve. The fastest technique for stabilizing in this area is to set up the aircraft near the desired speed and altitude using gross techniques. Hold altitude precisely ($\pm$ 5 feet) by the methods described above, and change power in small increments until the aircraft stabilizes on the desired speed. A special modification of this procedure for obtaining a point at maximum speed is to climb above the desired altitude and dive so that the altitude is reached at a speed slightly above V_{max}. Then if altitude is held precisely the speed will rapidly settle to V_{max}.

The speed range below approximately 180K on the above chart is the back side of the power curve. The technique outlined above will not be satisfactory in this speed range for the following reason. Suppose that it is desired to stabilize the aircraft at point A. If the pilot guesses his power setting of 92 % correctly but stabilizes first at a slightly higher speed such as that of point B, he will have more power than necessary to maintain stabilized level flight. If altitude is held precisely the aircraft will therefore accelerate steadily until it stabilizes at point D on the front side of the power curve.

Conversely, suppose the pilot sets the aircraft on the desired speed and altitude but does not apply sufficient power placing the aircraft at point C. If he holds altitude and does not change power the aircraft will have insufficient power to maintain speed and will promptly decelerate to the stall. If the pilot applies power to stop the deceleration the aircraft can be stabilized at point E, but now it is necessary to reduce power again to stabilize at point A.

In fact, even if the pilot were to select exactly the correct power setting and were to place the aircraft exactly at point A, it would not stabilize there. If altitude were held constant and power setting remained constant the aircraft, by virtue of fuel consumption, would over a period of time, become lighter. Its drag would then decrease slightly and the aircraft would begin to accelerate. After accelerating to a higher speed the power required would become less and it would accelerate faster. In a period of time it would arrive and stabilize at point D.

For this reason it is necessary to use a special "back side technique." The method is to establish the aim airspeed exactly ($\pm$ 1/4 knot or less) and hold it by means of small attitude corrections. Altitude is then corrected and maintained by means of power. If airspeed is held constant it will become evident in a short time whether the aircraft is climbing or descending. Altitude can then be readjusted and a second attempt made with a corrected power setting.

For some aircraft there exists a speed below which zero rate of descent cannot be maintained at full power. This speed may or may not coincide with the aerodynamic stall speed and is known as V_{min} for stabilized level flight. Determining this speed presents a special problem. First, the pilot should guess at V_{min}, attain and hold this speed at full power. If the aircraft starts to climb, an indication that the speed is above V_{min}, the power should be reduced and a new attempt made at a slower speed. On the other hand, if the aircraft starts a descent while maintaining this aim airspeed at full power the speed was too low. In this case, the aircraft must be dived to gain airspeed in order to regain lost altitude. The next attempt should be made at a higher speed.

The area from approximately 180K to 230K on the chart also requires special attention. In this area a very small power change will produce a large change in speed, but the speed will change very slowly. It is therefore found more practical to use the back side technique to ensure that speed does not change after the aircraft is supposedly stabilized.

The back side technique described

above can be used successfully at any speed up to V_{max}, but front side technique is the quicker of the two when it can be used. As the test pilot becomes more proficient at the back side method he will usually find himself employing it over more and more of the aircraft speed range.

Aircraft having a well-defined aerodynamic stall characteristic may have only a small section of the speed range on the back side of the curve, or even none at all. This is particularly true at low altitude where the bottom of the curve occurs at fairly low speed. At high altitudes nearly all jet aircraft will have a well-defined back side portion to the power curve.

TRIM: Since it has long been accepted that for stabilized flight conditions the properly trimmed aircraft will do a better job of flying alone than with the pilot on the controls, it is most important that the performance test pilot understand and apply proper trim techniques.

At the present time most performance flight tests require the aircraft to be in a stabilized condition. Therefore, whenever possible the aircraft should be trimmed for hands off flight. This will allow the pilot to perform his many duties of reading instruments, recording data, etc., without the aircraft wandering from the stabilized condition.

The proper trim technique is first obtain the proper attitude for a given stabilized airspeed using the attitude technique described above, then to relieve the stick forces with the trim tab until the aircraft will fly hands off at the stabilized airspeed. If the aircraft is not trimmed properly the pilot will first note an attitude change followed by a change in airspeed. In this case, the pilot should first obtain the proper attitude and airspeed before making a change in trim. Accurate trimming can be quickly accomplished if, when the controls are released, the pilot returns the aircraft to the original trimmed attitude before the airspeed can change. The performance test pilot should become extremely trim conscious.

list of abbreviations and symbols

A	Area
a	Velocity of Sound
a	Acceleration
AR	Aspect Ratio $= \frac{b^2}{S} = \frac{b}{c}$
b	Wing Span
bhp	Brake Horsepower
C	Constant
°C	Degrees Centigrade
C_D	Drag Coefficient
C_{D_i}	Induced Drag Coefficient
C_{D_p}	Parasite Drag Coefficient
C_L	Lift Coefficient
C_p	Pressure Coefficient $\Delta P_p / q_{c_{ic}}$
CAT	Carburetor Air Temperature
D	Drag
e	Oswald's Efficiency Factor
F	Thrust
F	Force
°F	Degrees Fahrenheit
FAT	Free Air Temperature
g	Acceleration due to Gravity
H	Height
ΔH_{ic}	Altimeter Instrument Correction
h_e	Specific Energy
ΔH_{pc}	Altimeter Position Error Correction
hp	Horsepower
K	Constant
K	Temperature Probe Recovery Factor
°K	Degrees Kelvin
K.E.	Kinetic Energy
L	Length
L	Lift

M	Mass	Δt	Time Differential
M	Mach Number or True Mach Number	T.O.	Takeoff
ΔM_{pc}	Mach Number Position Error Correction	TOD	Time of Day
MP	Manifold Pressure	V	Volume
N	Engine RPM	V	Velocity
n	Load Factor	ΔV_c	Airspeed Compressibility Correction
P	Pressure	V_e	Equivalent Airspeed
p	Power	V_g	Ground Speed
ΔP	Pressure Differential	ΔV_{ic}	Instrument Correction for Airspeed
psi	Pounds Per Square Inch	ΔV_{pc}	Airspeed Position Error Correction
psf	Pounds Per Square Foot	V_w	Wind Velocity
q	Dynamic Pressure or Impact Pressure	W	Weight
q_c	q Compressible Flow ($P_t - P_a$)	W_a	Air Flow
$q_{c_{ic}}$	q Corresponding to V_{ic} ($P_t - P_s$)	W_f	Fuel Flow
°R	Degrees Rankine	α(alpha)	Angle of Attack
R	Gas Constant	γ(gamma)	Specific Heat Ratio and Climb Angle
R	Radius	δ(delta)	P_a / P_o
RF	Range Factor	η(eta)	Efficiency
R_n	Reynold's Number	θ(theta)	T_a / T_o
R/C	Rate of Climb	μ(mu)	Coefficient of Viscosity
S	Area	π(pi)	3.1416
T	Temperature	ρ(rho)	Density
T_{ex}	Excess Thrust	σ(sigma)	ρ_a / ρ_o
t	Time	ω(omega)	Angular Velocity
ΔT	Change in Temperature		

SUBSCRIPTS

a	Ambient Conditions or Available
c	Calibrated
g	Gross
i	Indicated (Instrument Reading)
ic	Instrument Corrected
n	Net
o	Standard Day – Sea Level Conditions (All Quantities with this Subscript are Constant)
r	Required
s	Standard Day Conditions
t	Test Day Conditions or True Values
t_2	Conditions at Compressor Inlet

CHAPTER I

CALIBRATION TESTS

1.1 PACER TEST

The pacer test is one of several methods which may be used to determine the position error of the pitot static system. It is simple to fly and the data is simple to reduce. It has the disadvantage of requiring a specially calibrated pacer aircraft the position errors of which are known. Accuracy of the results are no better than the accuracy with which the pacer aircraft has been calibrated.

In general, the test consists of flying the test aircraft in approximately line abreast formation with the pacer aircraft and taking simultaneous airspeed and altimeter readings. A single flight will usually be organized to take data at a single altitude at various stabilized speeds throughout the speed range of the aircraft. Other flights may be included in the program to repeat the procedure at various altitudes throughout the altitude range. Since instrument error and position error are known for the pacer aircraft, the calibrated speed and altitude for any desired point can be found. These will be equal to the calibrated speed and altitude for the test aircraft and therefore the airspeed and altimeter position errors for the test aircraft can be found.

PRE-FLIGHT PREPARATION:

First, aim airspeeds must be selected and a data card prepared. Aim speeds should begin near minimum speed and increase in fairly even increments to near maximum speed, then decrease to near minimum. It is not necessary to use exactly even increments and in fact it is often desirable to obtain points closer together at low speeds where the data scatter is likely to be greater. Select even 10 knot or 5 knot increments for easy reading in flight. A sample data card arrangement is shown below:

Aim V_i	V_i	H_i	T_i	T.O.D.	F/C
120					
140					
160					
200					
etc.					

The free air temperature (T_i) is read in case a temperature probe calibration is desired. Fuel counter readings (F/C) are taken at a few data points for nW/δ computations. Time of day (TOD) is recorded at a few selected points to aid in data correlation.

An altitude should be selected for each flight which will be free of turbulence. Radio and hand signals for normal flight procedures, test procedures, and emergency procedures must be agreed upon, and finally all crew members of both aircraft must attend a briefing which covers:

a. Normal formation procedures.

b. Radio procedures and channels.

c. Test procedures and signals.

d. Emergency procedures and signals.

IN-FLIGHT TECHNIQUE:

It is not necessary for the pacer and test aircraft to be of the same type, nor is it necessary for the pacer aircraft to lead. It is usually easier for the more maneuverable aircraft to fly the wing position. If one or both of the aircraft have wing mounted pitot-static probes, the wing aircraft should be flown on the side which places the probe, or probes, out of all flow interference. Also the wing aircraft should fly line abreast of and level with the lead aircraft and with at least 1/2-span separation.

To stabilize at the same airspeed the wingman should align two points on the lead aircraft so that relative motion will readily be apparent. For accuracy, the two points should be as far apart as possible (usually one on the wing tip and one on the fuselage) and they should be almost at 90 degrees to the direction of flight.

At the test altitude the lead aircraft should level off and stabilize at the first aim airspeed. When the aircraft is completely stabilized and the wing aircraft is in position, the pilot (in the lead aircraft) transmits "on speed." When the wing aircraft is completely stabilized the pilot transmits "read." At this instant both pilots will read V_i, H_i, and T_i. The formation should then be opened up a safe distance while data are recorded.

An alternate procedure for signaling between aircraft is the use of hand signals. In this case, instead of the lead aircraft pilot transmitting, he will raise his hand and hold it where it can be seen by the wingman. When the wingman is fully stabilized, he will raise his hand then lower it as a signal to read. The remaining procedure is identical to that described above.

DATA REDUCTION OUTLINE:

Data reduction for this test is based upon the theory that if the two aircraft are flying at the same speed and altitude, their calibrated speeds and altitudes are the same.

Then $V_{c_{pacer}} = V_{c_{test}}$

and $H_{c_{pacer}} = H_{c_{test}}$

Thus $$\Delta V_{pc_{test}} = V_{c_{pacer}} - (V_i + \Delta V_{ic})_{test} \quad \text{(Eq 1.1)}$$

$$\Delta H_{pc_{test}} = H_{c_{pacer}} - (H_i + \Delta H_{ic})_{test} \quad \text{(Eq 1.2)}$$

Column	Symbol	Units	Description
1	$(V_i)_t$	knots	Test aircraft indicated airspeed
2	$(\Delta V_{ic})_t$	knots	From laboratory calibration
3	$(V_{ic})_t$	knots	
4	$(V_c)_p$	knots	Pacer indicated airspeed corrected for instrument error and position error
5	$(\Delta V_{pc})_t$	knots	$(V_c)_p - (V_{ic})_t$

The following steps provide an alternate method for determining position error through altitude readings. This section is superfluous and will normally be used only if the results of steps 1 through 5 are in doubt.

6	$(H_{ic})_t$	feet	Test altitude corrected for instrument error
7	$(H_c)_p$	feet	
8	$(\Delta H_{pc})_t$	feet	$(H_c)_p - (H_{ic})_t$
9	$(\Delta V_{pc})_t$	knots	Through conversion charts

1.2 TOWER FLYBY TEST

The tower flyby test is one of the more reliable methods of obtaining a pitot-static system calibration. While it is accurate for all speed ranges, local conditions usually restrict its use to subsonic speeds.

This test consists of flying the aircraft past a theodolite, usually located in a tower, at a stabilized altitude and airspeed. Passes are made at a number of selected speeds throughout the desired speed range and on each pass the actual

altitude of the aircraft is measured by the theodolite operator, using the relations below.

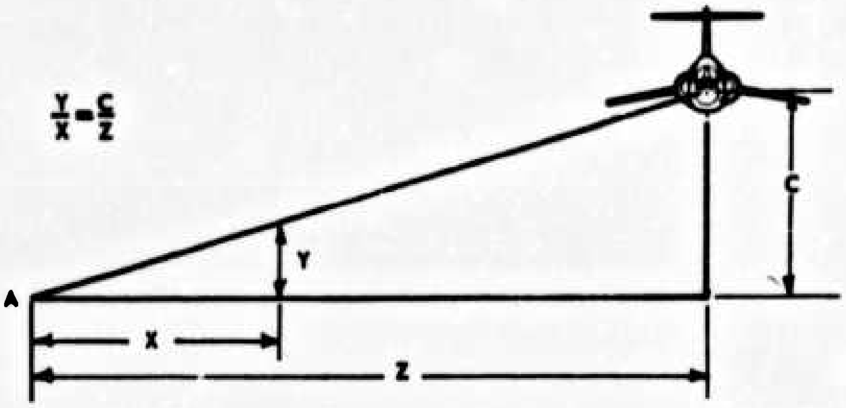


FIGURE 1.1

Normally a simple theodolite is used. It consists of a ring-shaped sighting eyepiece (location A, Figure 1.1), and a flat grid plate a known distance X from the eyepiece (location Y, Figure 1.1). The grid is marked off in inches with a zero line at the level of the eyepiece. By sighting through the eyepiece the aircraft can on each pass be lined up with a grid and a reading taken. Then by similar triangles the height of the aircraft is computed from

$$C = Z\frac{Y}{X}$$

The aircraft must maintain a position over a known course line in order that the distance Z will be known.

The above procedure determines the true height of the aircraft above ground level. It then becomes necessary to convert this to pressure altitude (H_c). Several methods are available, of which the simplest for data reduction purposes is to place an altimeter in the tower at theodolite eyepiece level. Then immediately after each pass this altimeter can be read and its instrument corrected value will be the pressure altitude of the eyepiece. The altimeter should be set at standard (29.92" Hg) setting. The assumption is made that if the aircraft flies by at 75 feet above eyepiece level, for example, then the correct pressure altitude is eyepiece pressure altitude plus 75 feet. This assumes that the pressure variation from eyepiece level to airplane level is standard which of course is not quite valid but the departure from standard over the small altitude increments concerned will produce only a minor error. This method produces a certain amount of scatter in the results, primarily because of hysteresis errors in the tower altimeter. Tapping the altimeter will help but will not remove the ambiguity entirely.

An alternate method which fixes the pressure altitude more accurately is to use weather station pressure altitude as a basis. The height of the theodolite above the weather station must be known so that the distance of the aircraft above the weather station can be determined. Weather station readings should be requested approximately every fifteen minutes during the flyby period so that the variation of pressure altitude with time of day can be plotted. A chart may be drawn up as in Figure 1.2.

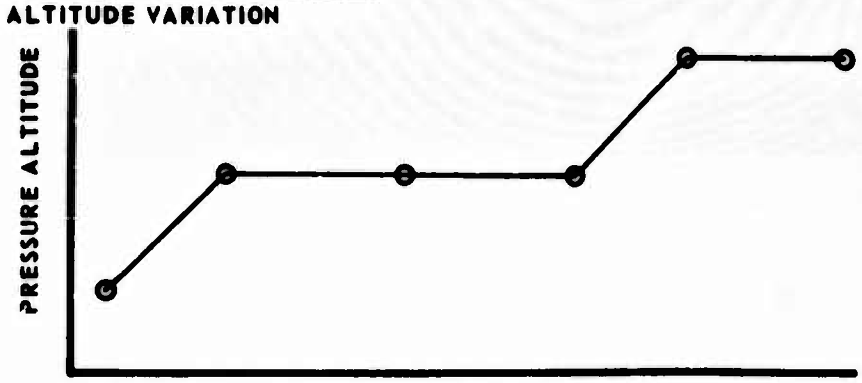


FIGURE 1.2

■ PRE-FLIGHT PREPARATION:

An airspeed schedule covering the full subsonic range of speeds should be selected. As in the pacer test, it may be desirable to use speeds spaced closer together in certain speed ranges. For example, the lower speed ranges will normally produce more scatter in test data and should be covered more throughly.

Hand-recorded data are suggested since they permit pilot judement. A normal data card layout would be:

Ground Block: TOD_____ H_i_____ Location___

Pass No.	Aim V_i	V_i	H_i	T_i	TOD	F/C
1	135					
2	150					
3	175					
	.					
	.					
	etc					

Ground Block: TOD_____ H_i_____ Location_____

Normally the high speed points are flown first, proceeding down to the low speeds as aircraft weight decreases. This procedure may be altered as necessary and is often reversed for training purposes. The tower operator should plan to record ambient temperature, theodolite reading and time of day for each pass.

A pre-flight briefing for all crew members participating in the tower flyby test is a necessity. This briefing should include call signs, radio procedures, pattern spacing, emergency procedures, and any special instructions concerning local area hazards. The tower operator should also attend the briefing.

Best results are obtained if smooth, stable air exists; therefore flight during the early morning hours is suggested.

■ IN-FLIGHT TECHNIQUE:

Before starting engines a ground block reading should be taken with the altimeter set to 29.92 inches Mercury. This consists of reading H_i (after tapping the guage) and time of day. After starting other ground block readings may be taken as desired at points on the aerodrome where the elevation is known.

After takeoff the tower flyby pattern is normally entered on the downwind leg, usually about 1000 feet above ground level. The aircraft should be thoroughly stabilized, the power setting noted, and the trim adjusted for hands-off flight. This trim setting should not be changed for the remainder of the pass.

The descent and turn to base and final will be one continuous turn with little or no power reduction if speed is high. At low speeds two 90° turns with an appreciable power reduction plus the use of speed brakes may be required. Stabilization on final will be easier if the aim airspeed is maintained through the base leg turns.

When rolled out on final on the flyby line the aircraft should be leveled off at an altitude slightly greater than one wing span above the ground and power applied to maintain the aim airspeed. Usually this will be slightly less than that used on downwind, but on each pass the pilot should note the speed at which stabilization occurs and make appropriate power adjustments on subsequent passes. Altitude should be maintained by a combination of external reference and altimeter cross-check. As the aircraft approaches the tower all corrections, including power, must cease so that it will pass in a fully stabilized flight condition.

For high speed points power control is simple. This consists of setting power early on final, and making one or two corrections in order to reach aim airspeed early. For low speed points, on the bottom portion and back side of the power curve, it will be necessary to use the back side technique. If a rate of descent, for example, occurs it will be necessary to increase power appreciably (usually 5% or more) until the aircraft reaches the desired altitude then reduce it to slightly higher than the original power setting. A rate of descent must be detected quickly and corrections made immediately as little maneuvering altitude exists. After two or three such attempts the altitude should be thoroughly stabilized and should remain so for the remainder of the pass. The pilot should be aware that there is a tendency to fly low on low speed passes, high on high speed passes and should correct his judgment accordingly.

As the tower is passed the altitude, airspeed, temperature, time of day, and fuel counter readings are noted. The aircraft should be placed in a slight climb and the data recorded before turning onto downwind for the next pass.

■ DATA REDUCTION OUTLINE:

If an altimeter is carried by the tower operator, the data reduction is as follows:

Column	Symbol	Units	Description
1	h	feet	Height of aircraft above eyepiece (from theodolite reading)
2	$(H_{ic})_{tower}$	feet	Instrument corrected value of tower altimeter reading
3	H_c	feet	Aircraft pressure altitude = $(H_{ic})_{tower} + h$
4	$(H_{ic})_{acft}$	feet	
5	ΔH_{pc}	feet	$H_c - H_{ic}$
6	V_{ic}	knots	Instrument corrected airspeed
7	ΔV_{pc}	knots	From conversion charts (if desired)

If it is desired to use weather station pressure altitude instead of the tower altimeter, the following steps can be used in place of columns 1 to 3.

Column	Symbol	Units	Description
8	Plot weather station readings vs time of day and join the points with a straight line as in Figure 1.2.		
9	TOD		Time of day at which pass was made
10	$(H_p)_{wx\ station}$	feet	Pressure altitude of weather station, read from the chart of column 8 at the time of day of Column 9
11	h'	feet	Height of aircraft above weather station, as determined from theodolite reading
12	H_c	feet	$(H_p)_{wx\ station} + h'$

Proceed with columns 4 to 7

An alternate solution using weather station pressure altitude is known as the "ΔH_{ic} ground block correction method." This method does not require that ΔH_{ic} be available. It assumes that ΔH_{ic} on each pass will be the same as when the aircraft is parked on the ramp. The assumption is valid if H_i does not change appreciably. If the aircraft in question has a large position error H_i may change 1000 feet or more, in which case ΔH_{ic} will also change and this method will not be valid. The procedure is as follows:

Column	Symbol	Units	Description
13	TOD		Time of day at which ground block reading was taken (for each ground block)
14	$(H_i)_{G.B.}$	feet	Ground block altitude
15	$(H_i)_{G.B.\ Corrected}$	feet	Ground block reading corrected to weather station level

(Example: If the ramp where the ground block is taken is 20 feet below the weather station and the aircraft altimeter is 5 feet above the level of the ramp, then $(H_i)_{G.B.\ Corrected}$ is ground block altitude plus 15 feet. $(H_i)_{G.B.\ Corrected}$ is what the altimeter would have read if it had been at weather station altitude.

16 Plot $(H_i)_{G.B.\ Corrected}$ on the plot of pressure altitude variation. The difference between $(H_i)_{G.B.\ Corrected}$ and weather station pressure altitude should approximately equal the laboratory value of ΔH_{ic}. (See Figure 1.3.)

PRESSURE ALTITUDE VARIATION

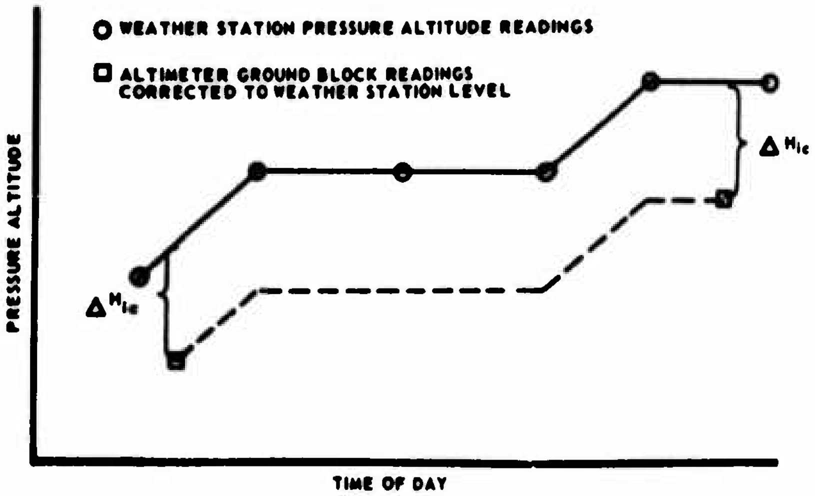


FIGURE 1.3

Column	Symbol	Units	Description
17	Draw a curve through the corrected ground block reading parallel to the pressure altitude curve (dashed line of Figure 1.3)		
18	$H_{i_{Base\ line}}$	feet	Read from the dashed line described above at the time of day at which the pass was made
19	h'	feet	Height of aircraft above weather station
20	$H_{c_{Base\ line}}$	feet	$H_{i_{Base\ line}} + h'$ (This is what the altimeter would have read on the pass if there had been no position error)
21	H_i	feet	Read directly from data card (no instrument correction)
22	ΔH_{pc}	feet	$H_{c_{Base\ line}} - H_i$

Proceed with columns 6 and 7

If the position error variation is large, laboratory values of ΔH_{ic} may be applied to $(H_i)_{G.B.\ Corrected}$ and to H_i on each pass (Columns 15 and 21).

If no laboratory calibrations or weather station data are available, a value for ΔH_{pc} can be obtained by simply assuming that pressure altitude variation at station level is linear. Then ground block readings before and after flight can be joined by a straight line which can be used as the base line.

If the position error curve is in doubt or if the laboratory value of ΔH_{ic} is in doubt, solutions by any two or all three of the above methods may be used and compared. The results may indicate what systems, if any, are faulty.

1.3 LOW ALTITUDE SPEED COURSE

This test yields airspeed position error directly from which other forms of the pitot-static error can be found. It consists of flying the aircraft over a measured course at low altitude (100 - 300 ft) and constant speed. Timing the flight gives the ground speed, from which true airspeed can be computed.

Wind will affect the ground speed computation on any one pass. However, it is possible to completely eliminate the effect of a headwind or tailwind component by flying each point in both directions, computing the ground speed for each pass and averaging the two. Note that speeds, not times, must be averaged to eliminate the error. The effect of a cross-wind component can be eliminated by flying a

heading parallel to the course and allowing the aircraft to drift. Wind effects can be eliminated in this manner only if the wind is constant in speed and direction and it is therefore advisable to fly the test when winds are light so that variations will be small.

This test is more effective for low speed aircraft, usually below 200 knots. At high speeds the time across the course becomes very short and minor timing errors become significant. Speeds should be chosen to cover the low-speed portion of the flight regime, with an allowance for safety. If a temperature probe calibration is being performed, the first point should be repeated at the end to pro-rate any temperature change.

PRE-FLIGHT PREPARATION:

A satisfactory data card for hand recording is as follows:

Run Direction	Aim V_i	H_i	T_i	Time	T.O.D.
S - N	200				
N - S	200				
S - N	180				
N - S	180				
	.				
	.				
	.				
	etc.				

If more than one aircraft will be on the course at one time it is important that speeds and procedures be correlated before the flight. Usually all aircraft will maintain the right side of the course.

Equipment required for the test are the data card and an accurate stop watch.

IN-FLIGHT TECHNIQUES:

The pilot must be familiar with the course so that he can fly it by proceeding from landmark to landmark at low altitude. The pattern is entered at about 500 feet above terrain at one end and a smooth descent to the 100 to 300 foot level is made. The aircraft is stabilized at the aim airspeed well before reaching the measured portion of the course and is precisely trimmed.

When the starting marker of the measured course is reached the stop watch is started. If airspeed and altitude are not precisely held, then a mental note must be made of their average values. If a temperature probe calibration is required, indicated temperature should be noted at the midpoint of the course and remembered.

At the end of the course the aircraft is placed in a slight climb and data recorded. The aircraft should not be trimmed as it is turned back for a second pass. It is important that the second pass be made at precisely the same speed as the first.

Back side points will present a special problem in that a very slight descent could result in the aircraft striking the ground before reaching the end of the course. A slight power change during the run may not completely invalidate the result but should be avoided if at all possible.

DATA REDUCTION PROCEDURES:

Since the test produces a true airspeed rather than calibrated values it is necessary to correct for density effects. One way of doing this is to find the position error in terms of Mach number, which depends only on calibrated speed and altitude (i.e., is independent of density). The data reduction is as follows:

Column	Symbol	Units	Description
1	V_g	ft/sec	Ground speed for each pass
2	V_t	knots	(Average value of V_g from two runs) x 1.6889
3	T_a	°C	From weather station or from $T_a = T_{ic} + K \frac{V_t^2}{7592}$

4	M	From V_t and T_a
5	V_{ic}	Average for two runs
6	H_{ic}	Average for two runs
7	M_{ic}	From charts at V_{ic} and H_{ic}
8	ΔM_{pc}	$M - M_{ic}$
9	$\Delta V_{pc}, \Delta H_{pc}$ etc	From charts as desired

The above data reduction will be equally applicable to a high altitude speed course such as photo theodolite or radar courses. Values of ambient temperature and pressure altitude will normally be obtained by a weather balloon at the time the course is being run. Such a method can be used for any speed range including supersonic.

The method may be inconvenient if available charts do not include small values of Mach number. An alternate solution is available using the relation

$$\Delta V_{pc} = V_t\sqrt{\sigma} - V_{ic} + \Delta V_c$$

The "compressibility correction" ΔV_c is less than 1/2 knot below 300 knots at 3000 feet, and thus can be ignored. The data reduction is then:

Column	Symbol	Units	Description
1, 2, 3			As previously described
4	H_{ic}	feet	Average value
5	σ	-	From H_{ic} and T_a
6	V_e	knots	$V_t\sqrt{\sigma}$
7	V_{ic}	knots	Average value
8	ΔV_{pc}	knots	$V_e - V_{ic} = V_c - V_{ic}$
9	$\Delta H_{pc}, \Delta M_{pc}$		As desired

1.4 SMOKE TRAIL TEST

For this test, a calibrated pacer aircraft capable of laying a smoke trail is required. The test aircraft is flown level with the smoke trail laid down by the pacer at a predetermined altitude and simultaneous airspeed and altimeter recordings are made. Usually the test will be performed by accelerating and decelerating along the smoke trail. For this reason, a photo-recorder is necessary unless the acceleration is extremely slow. If calibrations are desired in the transonic region, a photo-recorder becomes a necessity.

The assumption is made that the smoke does not change altitude. If this assumption is not valid the results will still be useful in defining the shape, but not the magnitude, of the position error curve.

It is possible, by means of the smoke trail test, to calibrate an aircraft pitot-static system over its entire speed range, including the supersonic range.

PRE-FLIGHT PREPARATION:

If a photo-recorder is used, it is necessary to record only run direction, camera counter numbers and time of day for correlation with the pacer aircraft. A thorough flight briefing covering speeds, altitudes, entry and departure procedures, signals, etc., should be attended by pilots of both aircraft. Radio communication is a necessity during flight.

IN-FLIGHT TECHNIQUES:

The pacer aircraft should stabilize at the desired altitude, at a speed for which its position error is well defined. The test aircraft should move back until the smoke trail begins to become poorly defined. It is desirable to have the pacer aircraft emit puffs of smoke at about 10-second intervals while this is done.

The pilot of the test aircraft then calls

for smoke and begins accelerating along the trail. Data should be taken at intervals as desired, with an event recorder used to indicate when the aircraft is level with the smoke and when it is not. If a Mach jump occurs, it is advisable to record data at CINE speed during the rapid changes. If the test aircraft's maximum speed is less than the pacer's, it is possible to perform accelerating and decelerating runs on one pass. Otherwise, the test aircraft will pass the pacer on the accelerating runs and must perform S - turns to reduce speed for another pass in another speed range. For high speed decelerating runs, an effective method is to climb above and descend to the smoke trail from several miles behind the pacer and commence deceleration as soon as the smoke trail level is reached.

To conserve smoke fluid, it is effective to have the pacer aircraft emit smoke puffs at 2- to 5-second intervals rather than in a continuous stream.

DATA REDUCTION:

Column	Symbol	Units	Description
1	H_c	feet	From pacer aircraft readings for the desired run
2	V_{ic}	knots	Test data
3	H_{ic}	feet	Test data
4	ΔH_{pc}	feet	$H_c - H_{ic}$
5	ΔV_{pc}, ΔM_{pc} etc.		As desired

1.5 OTHER POSITION ERROR PRESENTATIONS

After a number of calibration tests are complete it may be desirable to produce final plots which define the pitot-static position error through the full speed range of the aircraft. Several approaches are available and the choice will depend on the characteristics of the airplane, the shape of the position error curve, and the anticipated uses of the final plot.

The first steps in combining the results of the various tests will be to convert all data taken in the low speed region into a single form of the position error, producing a combined plot such as Figure 1.4. The curve of the figure, rather than the individual data points, now becomes the basis for further calculations. Similarly all data taken in the high speed regions will be plotted on a chart such as Figure 1.5.

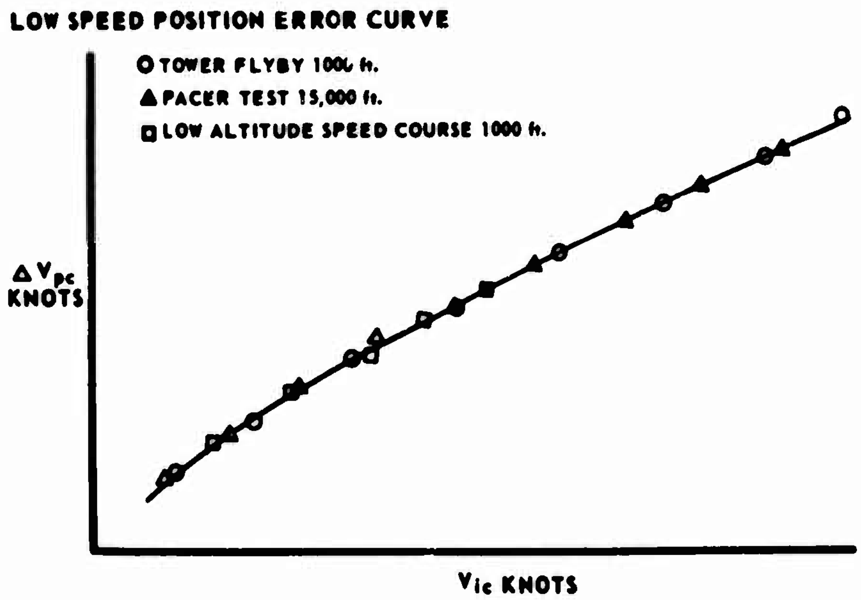


FIGURE 1.4

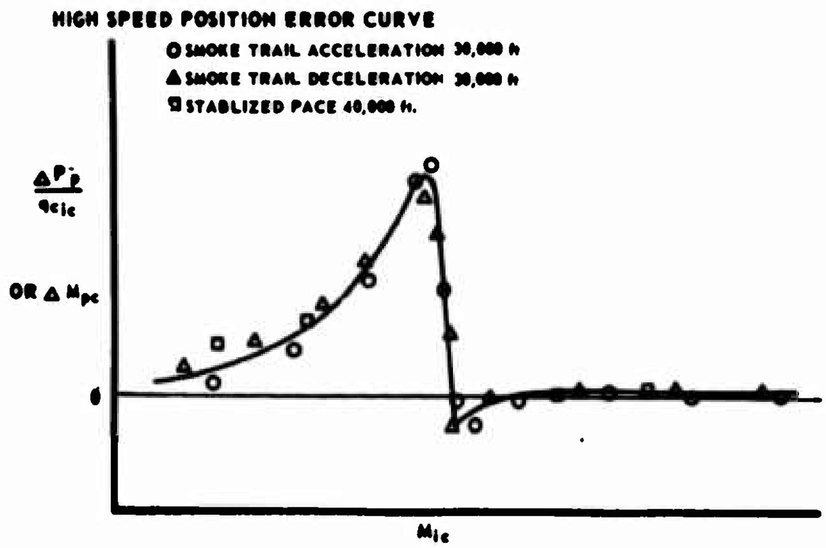


FIGURE 1.5

■ ΔV_{pc} VERSUS V_{ic}

The following data reduction outline presents the complete position error curve of the aircraft as ΔV_{pc} versus V_{ic}. Steps 1 to 11 may be eliminated if the gross weight change of the aircraft is small. In this case Figure 1.4 will be adequate in the low speed region.

LOW SPEED REGION

Column	Symbol	Units	Description
1	V_{ic}	knots	Select values at intervals in the low speed range. Example: 100, 150, 200, 250, etc.
2	W	lb	Average aircraft weight at which Figure 1.4 was obtained
3	$C_{L_{ic}}$		$\frac{nW}{1/2\ \rho_o\ V_{ic}^2\ S} = \frac{295\ nW}{V_{ic}^2\ S}$ S = aircraft wing area (ft^2) ρ_o = S.L. standard density $\frac{slugs}{ft^3}$ V_{ic} = knots W = aircraft weight (lb) n = 1.0 for level flight
4	ΔV_{pc}	knots	From Figure 1.4 at V_{ic}
5	Plot ΔV_{pc} versus $C_{L_{ic}}$		

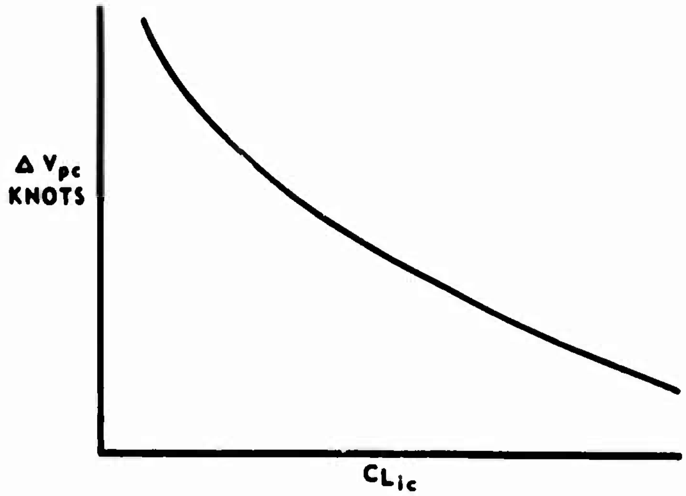


FIGURE 1.6

This curve is valid for all values of nW. To compute nW break-offs:

Column	Symbol	Units	Description
6	nW	lb	A selected value within the aircraft operating range
7	V_{ic}	knots	Selected values up to 0.9 M
8	$C_{L_{ic}}$		$\frac{295\ nW}{V_{ic}^2\ S}$ using nW from column 6
9	ΔV_{pc}	knots	From Figure 1.6 at $C_{L_{ic}}$
10	Plot ΔV_{pc} vs V_{ic} at this nW on the final plot (Figure 1.7)		
11	Select other values of nW and repeat steps 7 through 10		

HIGH SPEED REGION

Column	Symbol	Units	Description
12	H_{ic}	feet	A selected value
13	M_{ic}		Selected values from 0.5 to V_{max}
14	V_{ic}	knots	From charts at H_{ic}, M_{ic}

15	$\frac{\Delta P_p}{q_{c_{ic}}}$ (or ΔM_{pc})		Read from Figure 1.5 at M_{ic}
16	ΔV_{pc}	knots	Convert from $\frac{\Delta P_p}{q_{c_{ic}}}$ at H_{ic}
17	Select other altitudes and repeat steps 13 through 16		
18	Plot ΔV_{pc} versus V_{ic} at each altitude on Figure 1.7. Fair the curves of step 11 and step 18 into a smooth curve in the area of mixed effects (0.5 M to 0.9 M). Note that either ΔM_{pc} or $\frac{\Delta P_p}{q_{c_{ic}}}$ may be used as the source of data for high speed effects.		

COMPLETE POSITION ERROR CURVE

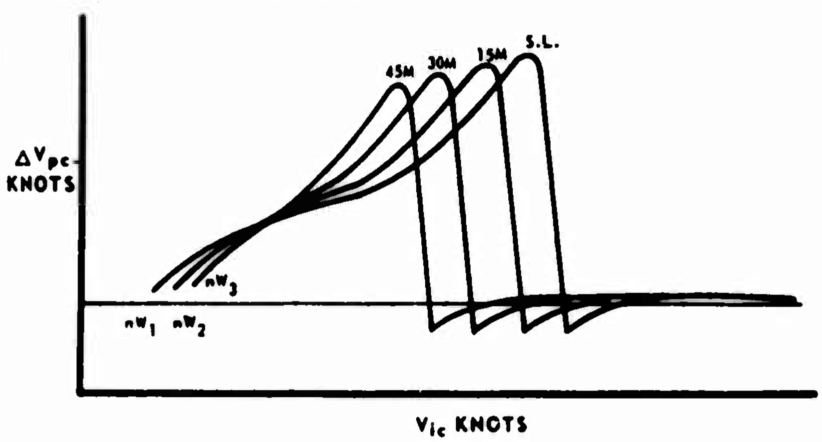


FIGURE 1.7

■ ΔM_{pc} VERSUS M_{ic} OR $\frac{\Delta P_p}{q_{c_{ic}}}$ VERSUS M_{ic}

The same basic charts (Figures 1.4 and 1.5) may be used as the source for these plots. Since the curves are being plotted against Mach number there will be only a single curve for all altitudes in the high speed region. For the low speed region the primary effect is altitude (δ). If the change in aircraft gross weight is a minor consideration, as it is in most fighter aircraft, the procedure is:

SMALL GROSS WEIGHT CHANGE

Column	Symbol	Units	Description
1	H_{ic}	feet	A selected value
2	M_{ic}		Selected values, V_{min} to 0.9M approximately
3	V_{ic}	knots	At M_{ic}, H_{ic}
4	ΔV_{pc}	knots	From Figure 1.4 at V_{ic}
5	ΔM_{pc} or $\frac{\Delta P_p}{q_{c_{ic}}}$		Convert from ΔV_{pc} at H_{ic}
6	Plot ΔM_{pc} versus M_{ic} at this altitude (see Figure 1.8)		
7	Select other altitudes and repeat steps 2 through 6		

COMPLETE POSITION ERROR CURVE

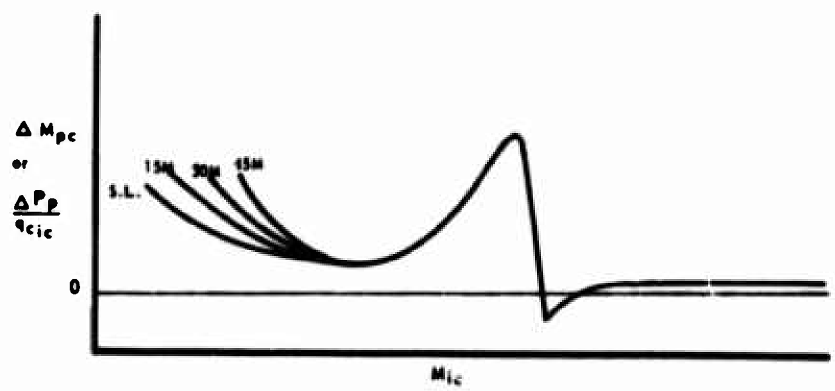


FIGURE 1.8

If aircraft gross weight change is a major factor the following procedure should be substituted for steps 1 through 7 preceding.

LARGE GROSS WEIGHT CHANGE

Column	Symbol	Units	Description
8	H_{ic}	feet	A selected value
9	$\frac{nW}{\delta}$	lb	A selected value corresponding to some desired weight range (heavy, medium, light) at δ corresponding to H_{ic}
10	M_{ic}		Selected values, V_{min} to 0.9
11	$C_{L_{ic}}$		$\frac{1}{1481\ M_{ic}^2\ S} \times \frac{nW}{\delta}$
12	ΔV_{pc}	knots	From Figure 1.6 at $C_{L_{ic}}$
13	ΔM_{pc} or $\frac{\Delta P_p}{q_{c_{ic}}}$		From V_{pc}, M_{ic}, H_{ic}

14 Plot ΔM_{pc} or $\frac{\Delta P_p}{q_{c_{ic}}}$ versus M_{ic} on the final chart (see Figure 1.9)

15 Select new values of H_{ic} throughout the operating range and repeat steps 9 through 14 for each.

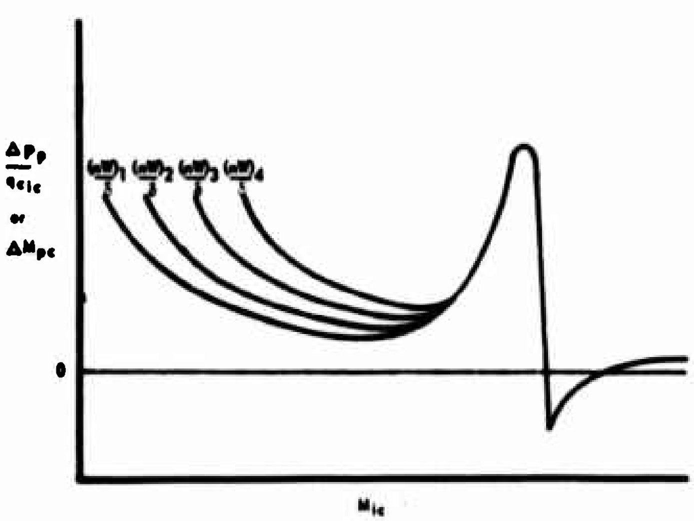


FIGURE 1.9

1.6 ALTITUDE POSITION ERROR

Altitude position error is fully defined by any of the complete position error curves described above or by a combination of Figures 1.4 and 1.5. However, to read altimeter position error in feet a conversion chart is necessary. A chart of ΔH_{pc} versus V_{ic} is therefore normally presented, along with one of the other forms described in Section 1.5. Such a chart can be obtained as follows:

LOW SPEED REGION

Column	Symbol	Units	Description
1	H_{ic}	feet	A selected value
2	V_{ic}	knots	Selected values through the low speed region
3	ΔV_{pc}	knots	From Figure 1.4 at V_{ic}
4	ΔH_{pc}	feet	Converted from ΔV_{pc}

5 Plot ΔH_{pc} versus V_{ic}

6 Select other values of H_{ic} and repeat steps 2 through 5 for each

If only the low speed or subsonic region is of interest a chart such as Figure 1.10 will result. If it is desired to cover the complete speed range, continue with steps 7 through 13.

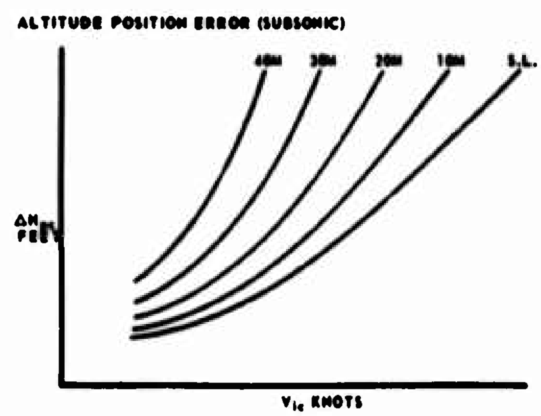


FIGURE 1.10

HIGH SPEED REGION

Column	Symbol	Units	Description
7	H_{ic}	feet	Same as step 1
8	M_{ic}		Selected values through the high speed range
9	ΔM_{pc}		From Figure 1.5 at M_{ic}
10	ΔH_{pc}	feet	Converted from ΔM_{pc}
11	V_{ic}	knots	At M_{ic}, H_{ic}
12	Plot ΔH_{pc} versus V_{ic}. Fair the curve into that of step 5		
13	Repeat steps 8 through 12 at the other selected values of H_{ic} from step 1.		

COMPLETE ALTITUDE POSITION ERROR

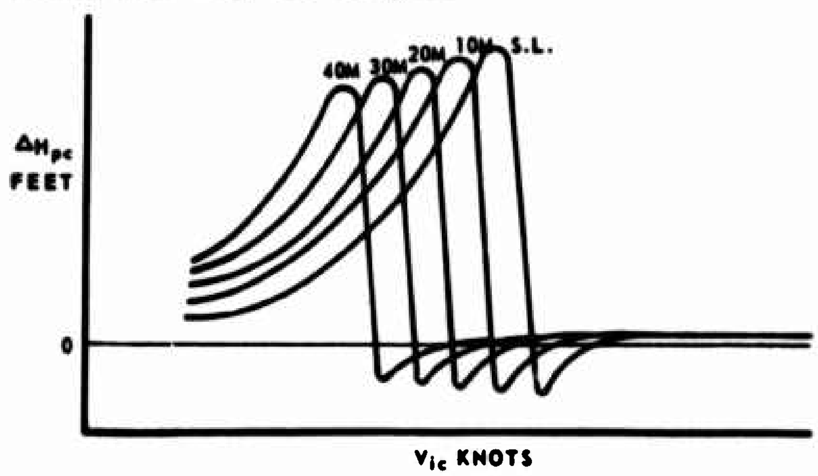


FIGURE 1.11

1.7 TEMPERATURE PROBE CALIBRATION

The temperature as indicated on tne free air temperature gauge is approximately total temperature and can be expressed by the equations:

$$\frac{T_{ic}}{T_a} = 1 + K\frac{M^2}{5}$$

or

$$T_{ic} = T_a + K\frac{V_t^2}{7592}$$

The purpose of calibrating the temperature probe is to determine the recovery factor (K) for the installation. K represents the percent of the theoretical flow energy or adiabatic temperature rise recovered by the probe.

Normally the temperature probe calibration test is flown in conjunction with some other test such as the Pacer, Low Altitude Speed Course, or Tower Flyby. Any test where the aircraft is flown at different speeds at a constant altitude in the same air mass while temperature and speed are recorded will suffice for calibrating the temperature probe.

The test is flown and reduced on the assumption that T_a is constant for all points. If this is not true, or unless corrections can be made, erroneous results will be obtained. Usually the procedure used to check for any change in ambient temperature during the period of time the test is flown is to re-fly the first point at the same airspeed. Any difference in the temperature readings is then due to a change in T_a and can be pro-rated on a time basis between all test points as shown in Figure 1.12.

CHANGE IN TEMPERATURE vs TOD

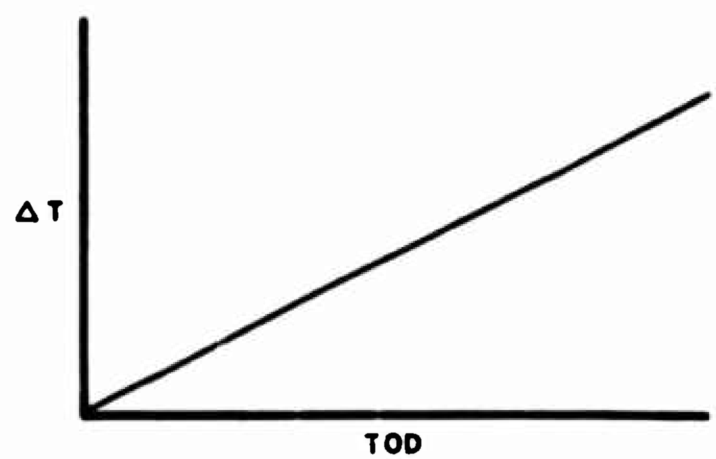


FIGURE 1.12

■ PRE-FLIGHT PREPARATION:

Pre-flight preparation for this test consists mainly of ensuring that V_i, H_i, T_i, and TOD are included in the items to be recorded. In addition, plans should be made to re-fly the first speed point (preferably a slow speed) at the end of the test so that any change in T_a can be pro-rated among all other points.

■ IN-FLIGHT TECHNIQUE:

No special techniques are required for this test - only those mentioned in other calibration tests need to be observed.

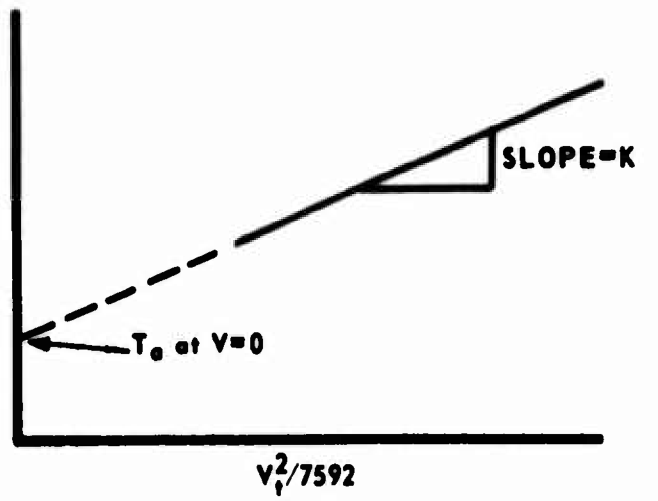
■ DATA REDUCTION OUTLINE:

Data reduction for the temperature probe test consists of determining the true speed or Mach number, correcting indicated temperature for instrument error and any change in ambient temperature, then plotting

$$T_{ic} \text{ vs } \frac{V_t^2}{7592} \text{ or } \frac{T_{ic}}{T_a} - 1 \text{ vs } \frac{M^2}{5}$$

as shown in Figure 1.13. The slope of the line on this plot is the recovery factor.

DETERMINATION OF RECOVERY FACTOR (K)



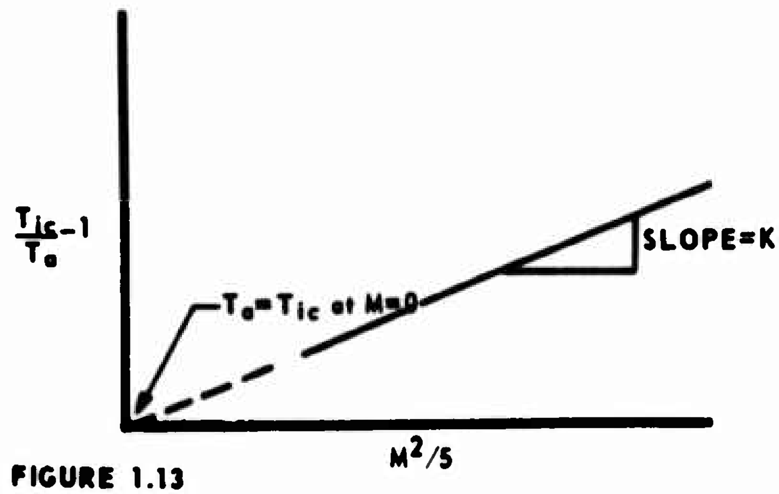


FIGURE 1.13

CHAPTER II

CLIMB TESTS

2.1 INTRODUCTION

This section includes those tests associated with determining the climb performance of an aircraft. Since the climb performance is dependent upon the speed schedule the aircraft flies, the first tests are designed to determine the speed schedule which the aircraft should fly. These tests are the sawtooth climb test for slower speed aircraft and level flight acceleration test for high speed aircraft.

Once the climb speed schedule has been determined, the actual climb performance of the aircraft is obtained by running check climbs to altitude. In general the same procedure is used for both reciprocating engine and jet aircraft. The factor which complicates the climb performance determination is the fact that all data must be corrected to standard day conditions. Test day performance is easily obtained but has no meaning if used to compare two aircraft flown on two different days. It is therefore necessary that sufficient data be recorded and proper techniques employed to reduce the results to that which would have been obtained if flown under ideal standard day conditions.

The major effect on aircraft performance is non-standard temperature. Other corrections such as non-standard weight, vertical wind gradient effects, and climb path acceleration are normally of lesser importance.

Due to the fact that large lag errors occur in the measurement of the free air temperature, the temperature as obtained from a weather station may be satisfactory and is sometimes more accurate than that obtained from the aircraft instrument when insufficient time has been allowed for stabilization.

2.2 SAWTOOTH CLIMB TEST

The sawtooth climb test is one method of obtaining the airspeed schedule for maximum rate of climb. Its name is derived from the barograph trace resulting from a series of short timed climbs through the same pressure altitude. This test provides little or no useful information on climb performance. It merely establishes the best airspeed at which to climb.

Essentially this test employs a trial and error method. A series of timed climbs are made at different speeds from a point below the test altitude to a point above it. Speeds are chosen to bracket the expected best climb speed of the aircraft.

Climbs are performed at the same power setting and aircraft configuration as will be used in the Check Climb. The altitude increment should be chosen such that the aircraft will traverse it in about one minute. Smaller time increments will introduce excessive scatter in the data.

The aircraft is first trimmed in the climb configuration while still well below the nominal altitude. Power is applied and final trim adjustments are made before reaching the lower limit of the altitude band being measured. The exact time of entering and leaving the altitude band is recorded by stopwatch or photo panel.

Upon emerging from the altitude increment, data is recorded and a 180° descending turn is initiated to bring the aircraft below the altitude band for another run. Trim settings should remain unchanged throughout the turn and descent, so that only a small trim adjustment will be required for the next run. As many

points as possible should be flown at each altitude. In addition, a full power unaccelerated minimum speed point and a maximum speed point should be obtained at the test altitude in order to complete the curve (see Figure 2.1). These two points should be flown at the beginning of the test so that weight corrections will not be necessary.

An effort should be made to remain in the same geographical location since the primary concern is the shape of the curve obtained rather than the magnitude. Any orographic lift acting on the aircraft will tend to remain constant if the aircraft remains in the same area relative to the ground.

For each altitude a standard data card should be prepared with the aim indicated airspeed (aim V_i) included for each point. Provision should be made for recording in flight actual V_i, Δtime, fuel counts, and either outside air temperature or time of day.

On the back of the data card a running plot of observed time to climb versus V_i should be kept, and before leaving the test altitude it should be examined for points that might need re-runs.

CORRECTIONS TO SAWTOOTH CLIMB DATA:

The theory by which corrections are applied is identical with that used in the Check Climb, as given in FTC-TIH 62-2005 Chapter II. If it is desired to correct sawtooth climb data to standard day conditions in order to substantiate check climb data, then all corrections must be applied. This will normally be done if a digital computer data reduction process is used. If only the <u>speed</u> for best rate of climb is desired, then many of these corrections may be ignored.

Major corrections which need not be included will be temperature correction and induced drag correction. Non-standard ambient temperature will produce a thrust change which will be almost constant throughout the airspeed range and will cause a vertical displacement of the points on Figure 2.1. The peak of the curve will still occur at essentially the same airspeed. The induced drag portion of the weight correction will be assumed to be insignificant for most aircraft. It need only be applied to aircraft with very high induced drag and/or a very large gross weight change.

Wind Corrections: The most effective method for coping with errors caused by vertical wind shear is to keep their magnitude as small as possible by:

1. Flying only in light, constant winds.

2. Flying on a heading of 90° to local winds.

3. Repeating some or all of the points on reciprocal headings.

The use of step (3) will result in a scatter of data points about a median line, serving to locate the line more accurately and to indicate the magnitude of wind errors present.

Acceleration Corrections: This is one of the two basic corrections applied to sawtooth climb data. It is shown by energy analysis to be the factor:

$$1 + \frac{V}{g} \frac{dV}{dH}$$

to be multiplied by the test rate of climb. This corrects the climb to zero acceleration. If the standard climb schedule departs appreciably from a constant V_t schedule, then the correction should be to the standard climb schedule value using an iteration process.

Inertia Correction: This portion of the weight correction consists of a quantity

$$\Delta R/C_2 = R/C_{corr}\left(\frac{W_t - W_s}{W_t}\right)$$

where R/C_{corr} is the test rate of climb with all other corrections applied. This quantity is to be added to the corrected rate of climb and has a large effect on

lateral displacement at the peak of the curve.

Note: Another method is available by which weight corrections can be applied if accurate fuel data is not available. (See end of this section.)

DATA REDUCTION:

Airspeed and altitude data should first be reduced to calibrated values. Ambient temperature should be found from T_{ic} using charts or from time of day and base weather data. Time should be calculated in minutes and apparent rate of climb dH/dT calculated.

The quantity $(1 + V\,dV/g\,dH)$ can be found using V_c and H_c by charts or by finding V_t from V_c and T_a and calculating $V_t/g \times \Delta V_t/\Delta H_c$. The quantity V_t in V_t/g should be found at the nominal test altitude.

The inertia portion only of the weight correction need be applied. The method previously described should be used. Weight corrections should be applied last.

A plot of corrected R/C vs V_c should be made with a smooth line joining the peaks of the curves to show the best climb speed schedule. (See Figure 2.1) The recommended climb speed is the greater of the two speeds and occurs at 99% of the maximum rate-of-climb value.

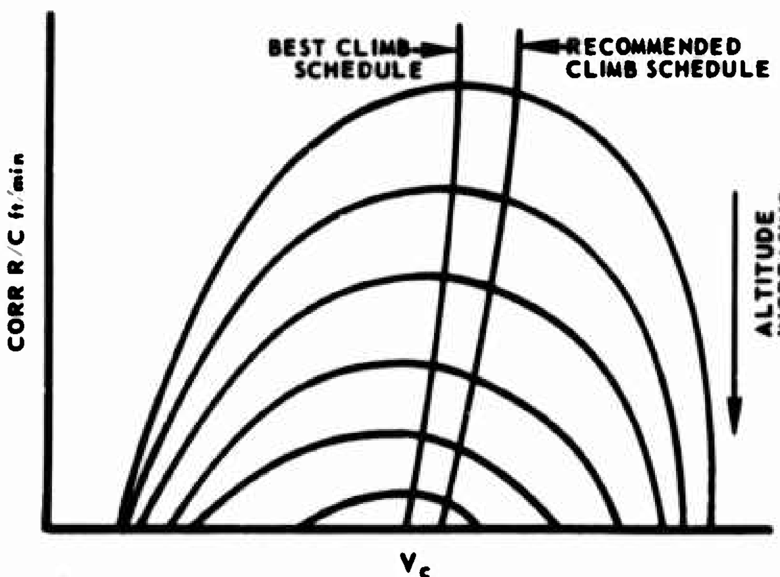


FIGURE 2.1

A final plot should be made of the climb schedule alone, on an upright altitude scale. Comparisons may be made on this plot if desired.

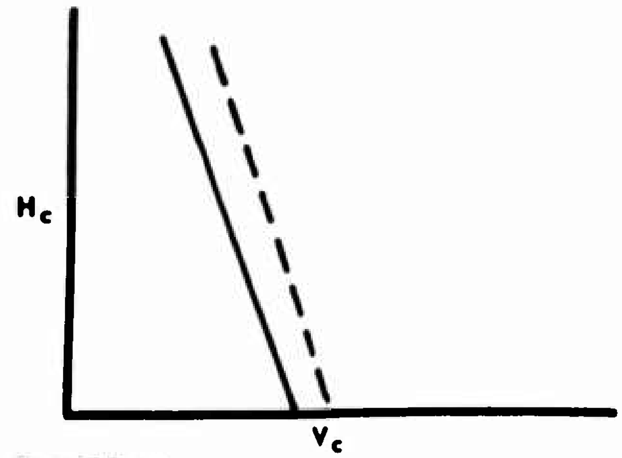


FIGURE 2.2

Alternate Weight Correction: Where complete instrumentation is not available, an alternate method of weight correction is available, based on a pro-rated rate of climb change with weight. While not perfectly accurate, it gives a method of correcting for both inertia and induced drag variation that is within the accuracy range of most test data.

For this method, points must be flown in order of decreasing or increasing airspeed. The high-speed point should be flown first, then points at decreasing speeds to the lowest speed, then the high speed point should be repeated. A working plot of R/C corrected for acceleration error vs V_{ic} must then be made, with both high speed points plotted. (See Figure 2.3)

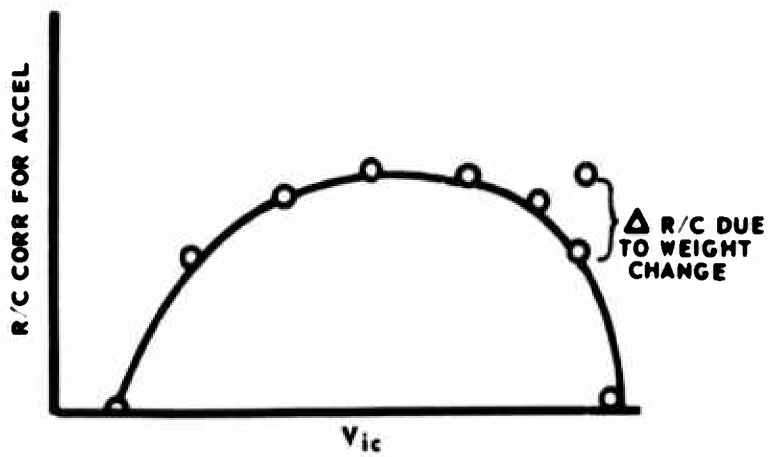


FIGURE 2.3

The $\Delta R/C$ due to weight change can now be found for the high speed point. The change in rate of climb is then assumed to be linear, and $\Delta R/C$ is calculated for other points. For example, if the first high-speed point were taken at 0800, the repeat point at 0900 hours, then for a point taken at 0830, the increase in rate of climb due to weight change will be ½ that at the high-speed point, and for a point taken at 0845 it will be 3/4, etc.

2.3 LEVEL FLIGHT ACCELERATION TEST

With the advent of high performance aircraft the performance envelope has greatly expanded and additional areas of investigation have become important. Higher wing loadings require higher climb speeds, and the acceleration from brake release to climb speed assumes greater importance. Supersonic capabilities result in a wide differential between best climb speed and maximum level flight speed, and level acceleration performance at altitude becomes important. And for most supersonic aircraft a supersonic climb speed schedule is of interest in addition to the familiar subsonic schedule.

The level flight acceleration test serves two purposes. It makes available acceleration time and fuel consumption data in level flight, and it may be used for determining climb speed schedules both subsonically and supersonically. The basis for the analysis of acceleration data is found in FTC-TIH 62-2005 Chapter IV, using a specific energy analysis.

METHOD:

Level flight accelerations from near minimum to near maximum airspeeds are normally flown at a variety of altitudes. As in the Sawtooth Climb test, power settings and aircraft configuration are those which will later be used in the check climb. Since a number of simultaneous readings are required and since data points are only a few seconds apart, recording must be mechanical, usually by photo panel. Values of indicated airspeed and time are the primary parameters. Fuel flow and free air temperature data are recorded and indicated altitude is included in order that errors caused by climb or descent may be corrected.

PRE-FLIGHT PREPARATION:

A data card should be drawn up and entries recorded in order that correlation between film and power settings, time of day, etc., may be facilitated. Required entries include:

RUN NUMBER, ALTITUDE, POWER SETTING (MIL OR MAX), RPM, COUNTER NUMBER (START AND FINISH), TIME OF DAY.

Any other desired parameters not included in the photopanel may be recorded as necessary.

The pilot should make himself familiar with altitude position error corrections of the aircraft so that he can plan for a slight indicated rate of climb or descent which will result in a nearly level flight path.

IN-FLIGHT TECHNIQUE:

The first step is to obtain a ground block reading. With the photopanel altimeter set at 29.92 inches take one or more shots on the ground and record the camera counter number. Pilot's and photopanel altimeters should remain set at 29.92 inches until the test is completed.

Upon reaching an area of smooth air at the desired altitude, the aircraft is stabilized at some medium speed and trimmed hands off. The aircraft is then slowed to just above minimum level flight speed without adjusting trim. The camera is turned on and adjusted to 1 frame every 2 to 5 seconds, and power is added to the desired climb setting. A smooth continuous rotation is begun and maintained by external reference, cross-checking the VSI occasionally to adjust the rotation rate. If the aircraft develops an unde-

sirable rate of climb, the rotation rate may be smoothly increased for a period of time. If a high rate of descent is obtained rotation may be slowed or stopped, but should not be reversed. If possible, the altimeter should be kept rotating in the same direction throughout the run in order to minimize hysteresis errors.

A critical point in the acceleration occurs when the aircraft passes through neutral trim. Care should be taken to progress smoothly from back stick pressure to forward without interrupting the rotation. Should accelerations be carried through the transonic region, both airspeed and altimeter indications will vary abruptly. A good technique is to continue a very slow rotation, allow airspeed and altitude readings to fluctuate, and reduce the rate of rotation slightly as speed increases. Some scatter will result in the data in any case because of lag and hysteresis effects.

When the rate of acceleration becomes quite small or when the desired Mach number is reached, data is recorded and power reduced. It may or may not be desired to record further data in decelerations or decelerating turns. The aircraft is then reduced to the speed for the next turn.

CORRECTIONS TO LEVEL FLIGHT ACCELERATION DATA:

As in the sawtooth climb, no corrections are required for temperature or for induced drag effects, if the data is being used only for obtaining the best climb speed schedule. In addition, weight corrections for inertia may also be omitted for hand reduction purposes, since (a) changes in gross weight of the aircraft do not greatly effect the climb speed of the aircraft, although they do effect the climb rate; and (b) the gross weight remains substantially constant through any given run.

For many aircraft with slow acceleration at high fuel flow rates these assumptions will not be valid. Also it will often be desired to obtain standard day acceleration figures as well as climb speed data. In these cases, the data will be changed to an equivalent rate of climb and corrections applied as in the check climb. Data may then be converted back to an acceleration basis.

As in the Sawtooth Climb, wind effects are avoided as much as possible by flying in still air. However, the direction of flight will make little difference in a level acceleration. If it is desired to check for a horizontal wind gradient a run may be repeated in the opposite direction and results compared.

DATA REDUCTION:

Counter number, airspeed, altitude, time, and temperature are recorded from the film and corrected to calibrated values. Airspeed and altitude data should be checked for direction of change and instrument corrections chosen from the appropriate curve. If altitude is increasing, for example, then correction should be chosen from the calibration curve marked UP.

True Mach number is determined from charts at V_c and H_c values. True velocity is then found, using charts or by

$$V_t = 38.967 \; M \sqrt{T_a} \quad \text{where } T_a \text{ in in } ^\circ K.$$

The value of V_t is changed to feet/second and the kinetic energy equivalent altitude

$$h_e = H + V_t^2/2g \quad \text{is then found.}$$

A delta time is calculated as the elapsed time from the start of the run to the point in question, and h_e and V_t are plotted versus time. A smooth curve is faired through the points. (Figure 2.4)

Approximately 15 equally spaced values of V_t are selected and the corresponding values of time are marked off. At each time the slope of h_e vs time is read, usually with a small mirror, and the results recorded. (Figure 2.5)

These values are now plotted vs V_t (Figure 2.6). This is in essence the same curve as that obtained by the sawtooth climb test. The best climb speed

schedule is a line joining the peaks of the curves for various altitudes.

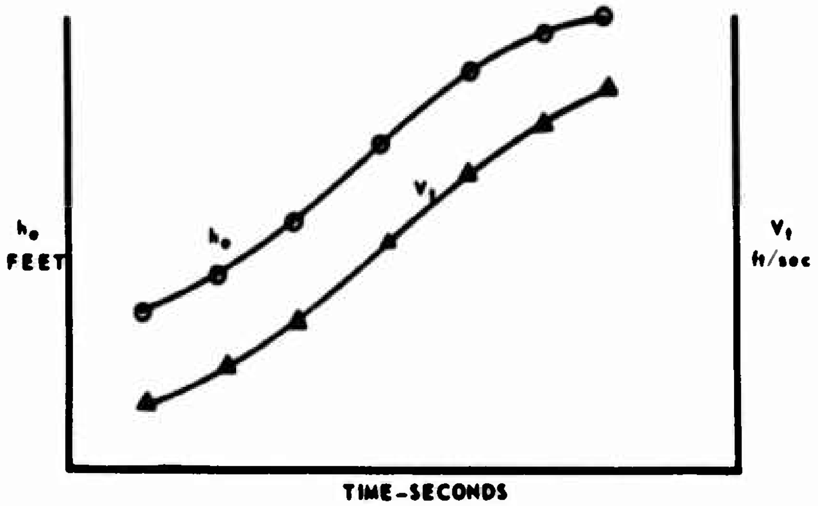


FIGURE 2.4

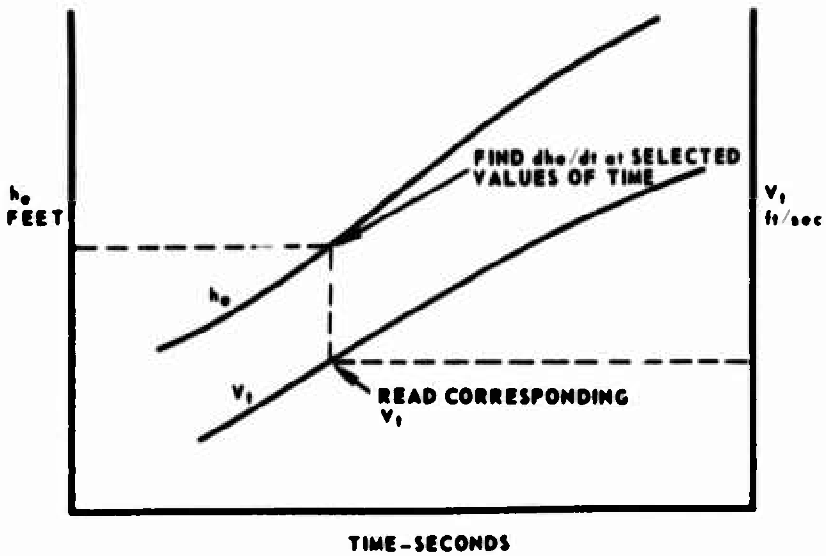


FIGURE 2.5

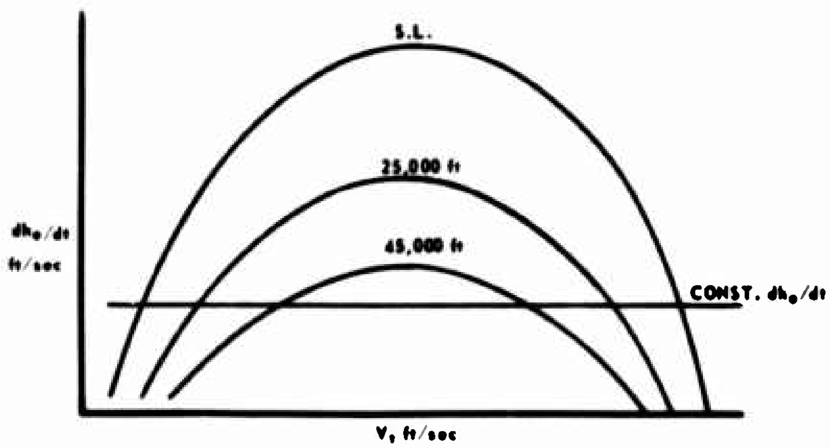


FIGURE 2.6

Lines of constant dhe/dt are drawn through the combined plots and these in turn are cross-plotted on a chart of H vs V_t (Figure 2.7). Values of V_t may be changed back to knots if desired.

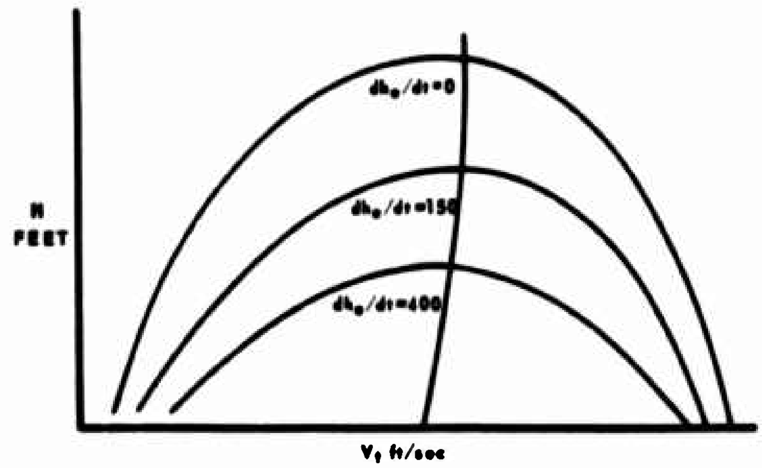


FIGURE 2.7

On this chart, the best unaccelerated climb speeds will also be found by a line joining the peaks of the curves. Lines of constant energy height h_e may also be drawn and the points where these are tangent to the lines of constant dhe/dt will define the optimum energy climb schedule (Figure 2.8).

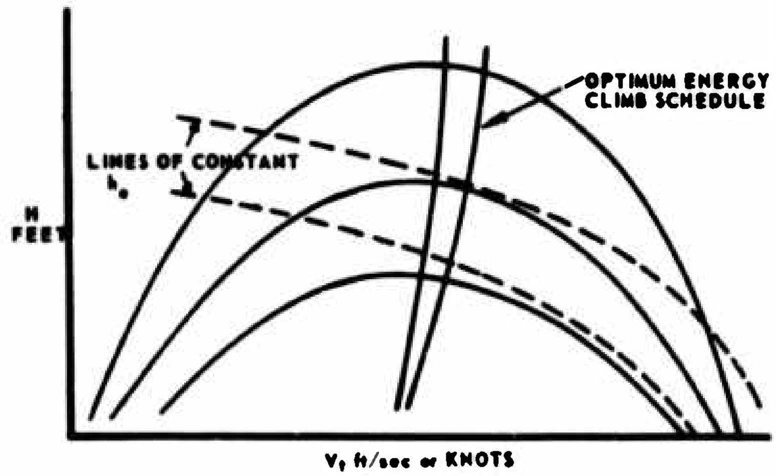


FIGURE 2.8

In practice, it is found to be much easier and almost as accurate to obtain an approximation of the optimum energy climb schedule. This is done by selecting an altitude 1 % below the peak of any dhe/dt curve and finding a point at this altitude on the high speed side. Joining

these points will give a climb schedule which, for the subsonic case, will usually agree with the optimum energy climb within the accuracy of data obtained. A supersonic climb schedule may also be found from this plot, although approximate methods may not be effective.

USES AND LIMITATIONS:

The level flight acceleration is the method used to obtain climb schedules for high performance aircraft chiefly because:

1. It is relatively easy to obtain good data at high rates of climb, and in fact the data accuracy improves at higher rates of climb.

2. Flying time required is much less than for the sawtooth climb. Usually one-half the time or less will cover the same speed and altitude range.

3. Wind gradient errors are smaller.

4. Acceleration data is obtained in conjunction with climb speed data.

5. Supersonic climb speed schedules can be found by this method.

Limitations in the method exist, in that:

1. At low climb rates the peaks of the curves are poorly defined. This limits the usefulness of the method for low performance aircraft, and for any aircraft as it approaches its service ceiling.

2. Data reduction processes are tedious if a hand reduction method is used. Even with a computer program considerable time is required for film reading and loadings.

3. Unless some form of mechanical recording device is available, data cannot be collected rapidly and accurately enough to be of value.

4. Because scatter is always fairly high, a single level acceleration is not reliable. From two to ten or more runs at any altitude are required to properly define the curve.

2.4 CHECK CLIMB TEST FOR JET AIRCRAFT

The check climb test is flown to evaluate the standard day climb performance of an aircraft in a specific configuration. The three main areas of investigation are:

1. Time to climb.

2. Distance travelled.

3. Fuel used.

In addition, data may be obtained on various engine parameters such as engine speed, exhaust gas temperature, engine pressure ratio, gross thrust, etc. These are useful to the analyst but are secondary to the three main parameters.

The general method is to climb the aircraft to just below the maximum ceiling while maintaining precisely a predetermined climb schedule. This schedule may be a best climb schedule as obtained by flight test, a schedule recommended by the manufacturer or some other schedule for which climb performance is of interest. Care should be taken to specify on each climb performance chart the schedule flown.

Data should be recorded at approximately equal increments of altitude, and should include time, speed, fuel used, temperature, and any other desired parameters. For most jet aircraft a mechanical recording means will be necessary to obtain simultaneous readings of the many parameters of interest.

PRE-FLIGHT PREPARATION:

The data card serves a double purpose: it provides the pilot with a list of aim speeds for each altitude, and is used to record pertinent data facts. The first portion of the data card is used to record fuel used to start and taxi, and fuel and time required to accelerate from brake release to climb schedule. A sample layout follows:

RUNWAY TEMPERATURE__________ PRESSURE ALTITUDE__________ WIND__________

TAXI AND TAKEOFF			
DATA POINT	TIME	FUEL COUNTS	CAMERA NUMBER
ENGINE START			
BRAKE RELEASE			
INTERCEPTING CLIMB SCHEDULE			

CHECK CLIMB				
H_i	V_i	CAMERA NO.	TIME OF DAY	OTHER DESIRED ITEM
4 000	370			
6 000	362			
8 000	354			
10 000	347			
ETC.	ETC.			

Aim speeds should be adjusted for instrument error and position error of both the airspeed indicator and altimeter. If the anticipated rate of climb is low, aim speeds should be presented every 1000 or 2000 feet with speeds to 1/2 knot. If the rate of climb is high, every 5000 feet is sufficient with speeds to the nearest knot. It may be advisable to decrease the interval to every 2500 feet or 2000 feet as the rate of climb decreases with altitude.

Prehaps the most difficult step in obtaining good check climb data is finding an area of satisfactory meteorological conditions. An area of smooth air, light winds, and stable temperature gradients from ground level to the aircraft's maximum ceiling is desirable. A survey balloon should be sent up before the flight for wind and temperature data and an area chosen where the climb can be performed at 90 degrees to the average wind direction.

Since aircraft gross weight and fuel density are extremely important, arrangements should be made, if possible, to weigh the aircraft, fully fueled, immediately prior to takeoff. In any case fuel samples from the tanks should be taken to obtain fuel temperature and density.

IN-FLIGHT TECHNIQUES:

Data on fuel used and time for taxi, takeoff, and acceleration to climb schedule should be taken whenever conditions permit. Upon reaching climb speed it is usually advisable to discontinue recording data and start afresh with the check climb entry.

Two basic methods for entering a check climb are available. In either case, the first step is to establish the aircraft in level flight as low as possible, consistent with safety, and on the climb heading. If an intervalometer is being used to record data it should be on and running <u>before</u> entering the climb. In a test program the interval will normally be adjusted to take readings every 500 feet or less during climbs. For training purposes, 1000 feet is satisfactory. If no intervalometer is available, trigger shots may be taken at desired altitudes.

If the rate of climb is high the best entry is usually achieved by first stabilizing in level flight with partial power at some speed below the scheduled climb speed. The aircraft should be trimmed for hands off flight. When all preparations are complete and the data recorder is running, power should be applied and as the climb speed is approached the aircraft should be rotated to intercept and maintain the climb schedule. If rotation is begun too early, the aircraft may climb several thousand feet before intercepting the desired schedule. On the other hand, if rotation is begun too late, the rate of rotation will be rapid and it will be diffi-

cult to avoid overshooting the desired pitch attitude.

If the rate of climb is fairly low a better entry can sometimes be achieved by stabilizing exactly on the aim speed about 1000 feet below entry altitude. When preparations are complete and the aircraft trimmed, the power should be advanced smoothly and the aircraft simultaneously rotated to maintain airspeed. As the desired power setting is reached, the rotation should be stopped, at which time the aircraft will be approximately established on the climb schedule.

During the climb the aircraft should be constantly trimmed for hands-off flight. The climb schedule should be maintained to within ½ knot where possible, taking care to keep a steady bleed rate. A rapid cross-check between external horizon and the airspeed indicator is required. If the pitch attitude is very steep it may be necessary to substitute the aircraft attitude indicator for the external horizon during initial portions of the climb.

During climbs, wind gradient effects will appear as sudden airspeed changes. If these affect the climb speed schedule, corrective action is to make small but immediate attitude correction. If the aircraft does not respond at once, another correction should be applied. The pilot should also be prepared to take instant corrective action as the wind gradient affect dies away resulting in a climb speed error in the opposite direction.

At high altitudes the problem of maintaining a precise speed schedule becomes difficult. A slight rate of change of indicated airspeed implies a much larger rate of change of kinetic energy. Therefore, any undesirable trend is difficult to stop with relatively ineffective aerodynamic controls. The best way to cope with this problem is to avoid it by a rapid cross-check, precise control, and constant attention to trim. If corrections do become necessary, care should be taken to avoid reversing the motion of the airspeed indicator because of the resulting hysteresis problem.

Upon completion of the climb, data recording should be shut off to conserve film and pertinent items such as time of day recorded.

If a digital computer data reduction process is being used, it is often advisable to record data at cine speed during entry to the climb and during the last few feet of climb. This provides data points at small intervals at each end. These can be discarded by the machine without the loss of valuable data.

DATA REDUCTION:

Data obtained during takeoff and acceleration to climb speed may be reduced as a level acceleration if desired. However, since it is influenced by many local conditions in the traffic pattern, it is often sufficient to merely make an estimate of time and fuel used based on uncorrected test results.

The hand data reduction outline following contains no correction for wind gradient. If wind effects cannot be avoided it is recommended that climbs be made on reciprocal headings and plotted on one chart.

Steps 1 through 7 of the data reduction are devoted to obtaining smoothed values of time, airspeed, and temperature. Slope readings should be taken at test values of H_c so that fuel counter readings etc. will occur in conjunction with other data.

Column	Symbol	Units	Description
1	H_c	feet	From test data
2	V_c	feet	From test data
3	T_{ic} (or T_a)	°K	Test data
4	Time	min or sec	Clock reading
5	Plot V_c and T_{ic} versus H_c (See Figure 2.9)		

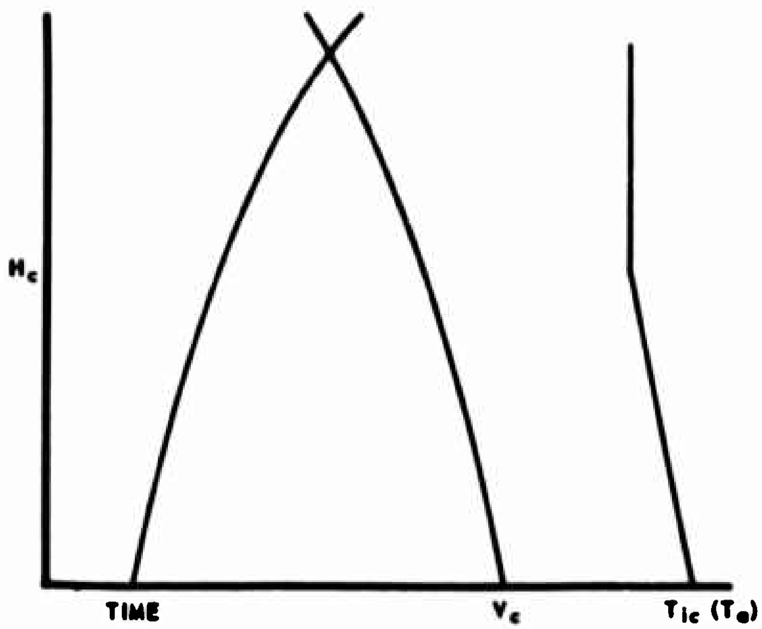


FIGURE 2.9

Column	Symbol	Units	Description
6	H_{c_t}	feet	Selected altitudes (About every 2000 feet for training purposes)
7	V_{c_t}	knots	Read from Figure 2.9 at H_{c_t}
8	M_t		From V_{c_t}, H_{c_t}
9	V_{t_s}	knots	From V_{c_s}, H_{c_t}
10	T_{a_t}	°K	Read from Figure 2.9 at H_{c_t} or from $T_a = \dfrac{T_{ic}}{1 + .2KM^2}$

Steps 11 through 14 provide a general approach to the thrust correction problem. This will vary with individual engine characteristics and some completely different approach may be required.

Column	Symbol	Units	Description
11	F.U.	lb or gal	Fuel used from test data
12	W_t	lb	Test gross weight
13	ΔF	lb	From manufacturer's charts at T_{a_s}, H_{c_t} and desired power setting.
14	$\Delta R/C$	ft/min	$\dfrac{V_{t_s} \; \Delta F}{W} \times 101.3$

Proceed to column 24

For T-33 data reduction using ARPS charts, the above section will be replaced by steps 15 through 23.

15	N_{ic_t}	RPM	Instrument corrected RPM
16	$N_{ic_t}/\sqrt{T_{a_t}}$		
17	$\left(Fn/\delta_{t_o}\right)_t$ or K_t	lb	From charts at $N/\sqrt{T_a}$ and M
18	$N_s\sqrt{T_{a_s}}$		N_s = 100 % RPM
19	$\left(F_n/\delta_{t_o}\right)_s$ or K_s	lb	From charts at $\dfrac{N_s}{\sqrt{T_{a_s}}}$ and M
20	ΔK	lb	$K_s - K_t$
21	W_t	lb	Test gross weight
22	$\dfrac{\Delta R/C}{\Delta K}$	$\dfrac{\text{ft}}{\text{lb-min}}$	From charts at H_{c_t}, M, and W_t per engine
23	$\Delta R/C_1$	ft/min	$\dfrac{\Delta R/C}{\Delta K} \times \Delta K$
24	R/C_2	ft/min	$R/C_t\sqrt{\dfrac{T_t}{T_s}} + \Delta R/C_1$

Steps 25 through 30 provide the acceleration correction. For hand reduction, it may be calculated at only a few points, plotted versus H_c, and intermediate values read from the plot.

25 From standard day charts and the desired climb schedule plot values of $\Delta V_{t_s} / \Delta H$ for all values of H_c

26	$\Delta V_{t_s} / \Delta H$	knots/ft	From the above plot at H_{c_t}
27	$(V_{t_t})_1$	knots	From M and T_a at H_{c_t} - 500 ft (Use Figure 2.9)
28	$(V_{t_t})_2$	knots	From M and T_a at H_{c_t} + 500 feet
29	$\Delta V_{t_t} / \Delta H$	knots/ft	$\frac{(V_{t_t})_1 - (V_{t_t})_2}{1000}$
30	$\Delta R/C_a$	ft/min	$\frac{V_{t_t}}{32.2} \times R/C_2 \times \frac{\Delta V_{t_t}}{\Delta H} \times (1.6889)^2$
31	R/C_3	ft/min	$R/C_2 + \Delta R/C_a$

Steps 32 through 39 determine the weight corrections. The induced drag portion (columns 34 through 39) may prove insignificant.

32	W_s	lb	Standard weight (from previous data or estimated)
33	$\Delta R/C_2$	ft/min	$R/C_3 \times \frac{W_t - W_s}{W_t}$
34	P_{a_t}	"Hg	From charts at H_{c_t}
35	b	ft	Aircraft wing span
36	e	-	Oswald's efficiency factor at M_t from wind tunnel data
37	$\sin \gamma$		$\frac{R/C_3}{V_{t_t} \times 101.3}$ γ = climb angle. If $\sin \gamma$ = 0.14 or less, let $\cos^2 \gamma$ = 1.0
38	$\cos^2 \gamma$		$1 - \sin^2 \gamma$
39	$\Delta R/C_3$	ft/min	$\frac{34.039 \sqrt{T_{a_t}}}{P_{a_2} b^2 M e} \left(\frac{W_t^2 - W_s^2}{W_t} \right) \cos^2 \gamma$
40	R/C_s	ft/min	$R/C_3 + \Delta R/C_2 + \Delta R/C_3$

All corrections to rate of climb are complete at this point. Steps 41 through 52 are for the correction of fuel flow data to standard day.

41 Plot fuel used against time and read slopes at the times for which data points were taken. (Figure 2.10)

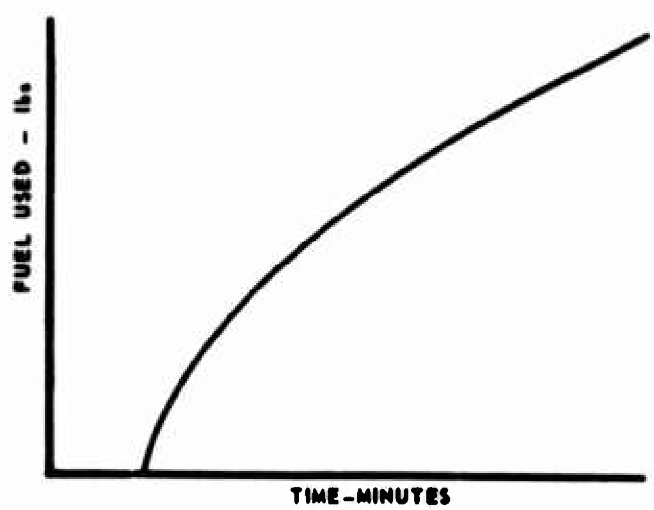


FIGURE 2.10

42 W_{f_t} lb/hr Slope from Figure 2.10 x 60

43 $\theta_{t_{2_t}}$ $\dfrac{T_{a_t}\ (1 + .2M^2)}{288}$

44 $\theta_{t_{2_s}}$ $\dfrac{T_{a_s}\ (1 + .2M^2)}{288}$

45 $P_{t_{o_t}}$ "Hg or psf $P_{a_t}\ (1 + .2M^2)^{3.5}$

46 $P_{t_{2_t}}$ "Hg or psf From $P_{t_{o_t}}$ and ram recovery plot.

47 $\delta_{t_{2_t}}$ - $\dfrac{P_{t_{2_t}}}{P_o\ (29.92 \text{ or } 2116.2)}$

48 Plot $\dfrac{W_{f_t}}{\delta_{t_{2_t}}\sqrt{\theta_{t_{2_t}}}}$ versus $\dfrac{N_{ic_t}}{\sqrt{\theta_{t_{2_t}}}}$ (Figure 2.11)

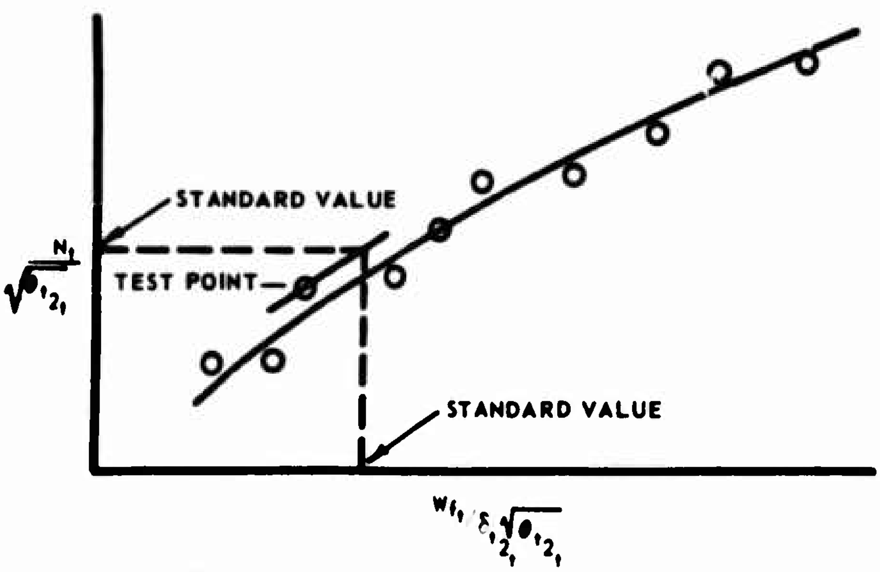


FIGURE 2.11

49 Draw a line through the test point parallel to the basic curve (Figure 2.11)

50 $\left(\dfrac{N}{\sqrt{\theta_{t_2}}}\right)_s$ RPM at 100 % RPM

51 $\left(\dfrac{W_f}{\delta_{t_2}\sqrt{\theta_{t_2}}}\right)_s$ lb/hr Read from Figure 2.11 as illustrated

52 W_{f_s} lb/hr $\left(\dfrac{W_f}{\delta_{t_2}\sqrt{\theta_{t_2}}}\right)_s \times \delta_{t_{2_t}}\sqrt{\theta_{t_{2_s}}}$

The remainder of the data reduction will be devoted to obtaining all desired parameters on a corrected time scale for the final plots.

53 Plot R/C_s, W_{f_s}, and V_{t_s} versus H_c (See Figure 2.12)

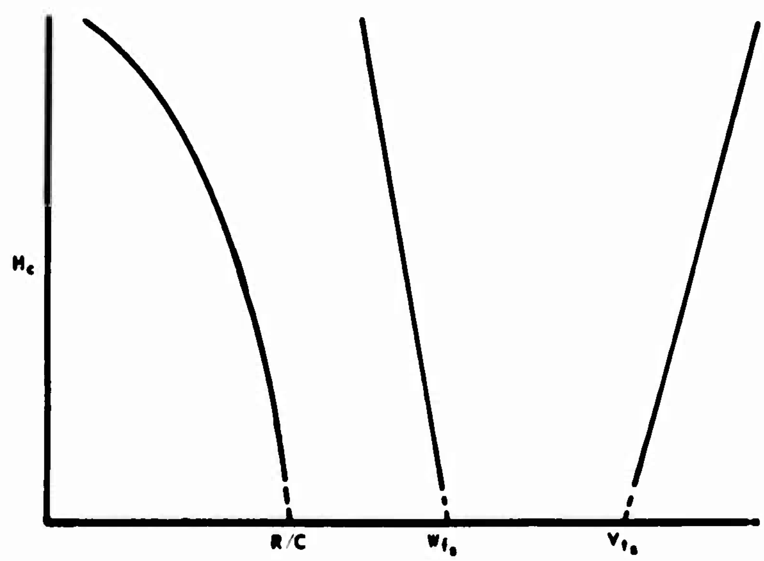


FIGURE 2.12

54 R/C_{avg} per 2000' ft/min From Figure 2.12. Example: For R/C_{avg} from S.L. to 2000 ft, read R/C at 1000 ft.

55	$\Delta T/C$	min	$\frac{2000}{R/C_{avg}}$
56	T/C	min	Summation of $\Delta T/C$
57	$W_{i_{avg}}$ /2000'	lb/hr	Similar to R/C_{avg}
58	$\Delta F\ U$	lb	$\frac{2000}{W_{f_{avg}}}$
59	Fuel Used	lb	Summation of $\Delta F\ U$
60	$V_{t_{avg}}$ /2000'	kt	Similar to R/C_{avg}
61	sine γ	-	$\frac{R/C_{avg}}{V_{t_{avg}} \times 101.3}$
62	cos γ		From sin γ
63	V_h	knots	$V_{t_{avg}} \times \cos\gamma$ (Horizontal velocity)
64	ΔNAM	n.m.	$\frac{V_h \times \Delta T/C}{60}$
65	NAMT	n.m.	Summation of ΔNAM. (Distance traveled)

66 Make a final plot (Figure 2.13). Use a double page or fold-out, or break into several charts.

CLIMB PERFORMANCE SUMMARY

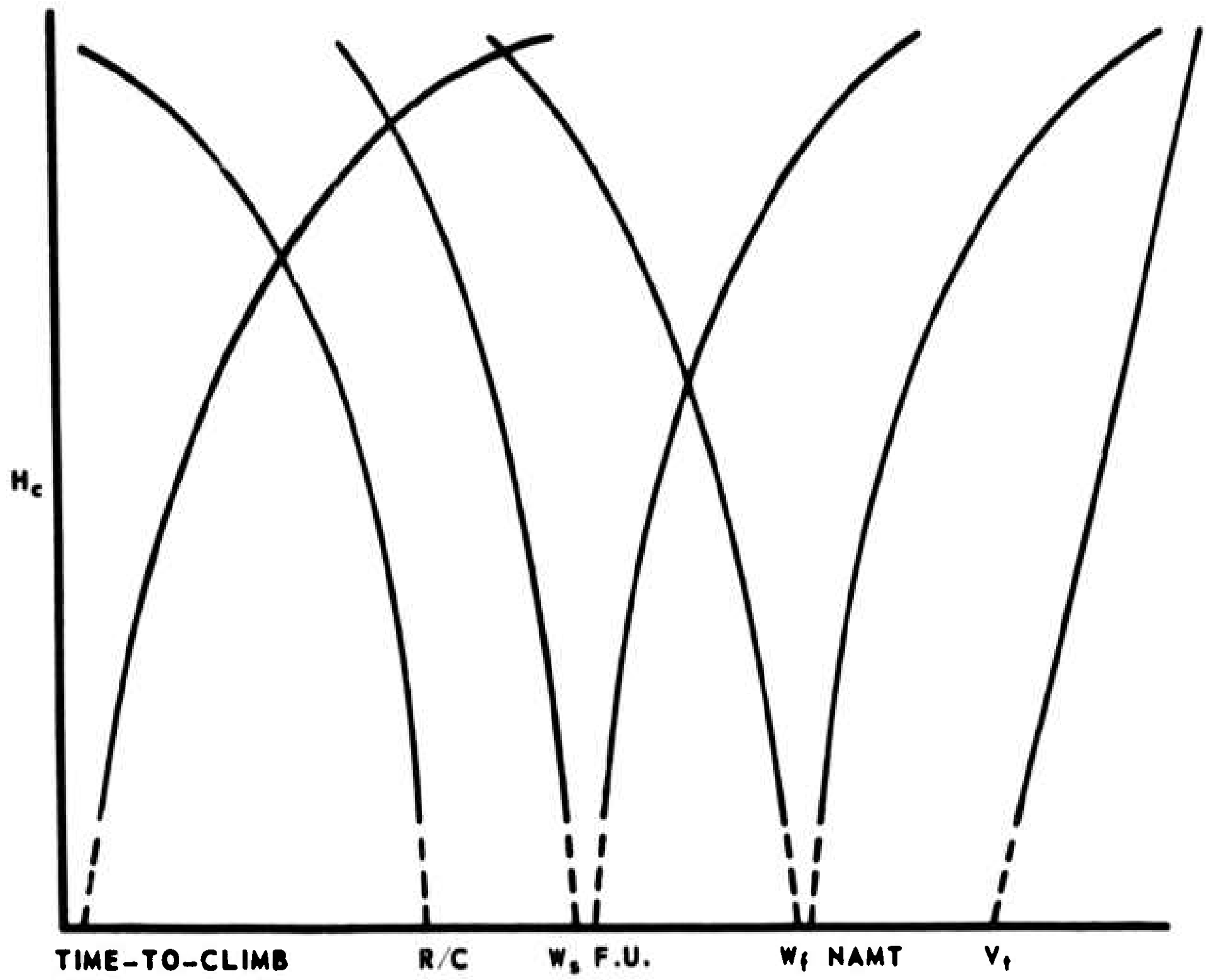


FIGURE 2.13

2.5 RECIPROCATING ENGINE CHECK CLIMB TEST

The primary purposes of the check climb test for reciprocating engine powered aircraft are identical with those of the jet check climb (Section 2.4). Minor items on which data is obtained at each altitude include available manifold pressure and brake horsepower.

Pre-flight preparation and in-flight techniques parallel closely those described in the jet check climb. An additional complication at lower altitude arises from the necessity of maintaining a predetermined manifold pressure with the throttle. A satisfactory method is to set the manifold pressure to approximately 1/2 inch Hg above the desired, then readjust every 1000 feet until critical altitude is reached.

If hand recording is to be used data should be recorded every 1000 feet if possible. The following items should be recorded:

Actual V_i	RPM	M.P.	Carb Air Temp
Free Air Temp	Fuel Count	Time	

The same precautions used to obtain accurate fuel consumption and gross weight data for jet aircraft should be observed.

DATA REDUCTION :

With the exception of the thrust correction, all data reduction methods used in the jet check climb can be applied directly to propeller powered aircraft. The acceleration correction and the induced drag correction will often be small for this type of aircraft in which case they may be ignored.

The thrust correction equations of columns 11 through 23 in the jet check climb data reduction should be replaced by the following:

Column	Symbol	Units	Description
11	N_{ic}	RPM	Instrument corrected tachometer reading
12	MP_{ic}	in. Hg	From manifold pressure gauge reading
13	BHP_{chart}		From engine manufacturer's charts at H_c, N_{ic}, MP_{ic}, and a standard carburetor air temperature CAT_s (°K)
14	CAT_{ic}	°K	
15	BHP_t		$BHP_{chart}\sqrt{\frac{CAT_s}{CAT_{ic}}}$
16	T_{a_s}	°K	From atmosphere charts at H_c
17	ΔT	°K	Change in ambient temperature, assumed equal to change in carb air temp, from $T_{a_s} - T_{a_t}$
18	C		A correction constant from engine manufacturer's charts
19	MP_s	in. Hg	$MP_{ic} \times (1 + C\ \Delta T)$
20	BHP_s		From charts at MP_s, H_c, N_s, CAT_s. N_s should equal N_{ic}
21	η		Propeller efficiency, from propeller manufacturer's charts.

22 W_t lb Test gross weight

23 $\Delta R/C_1$ ft/min $\frac{33\,000}{W_t}\left(BHP_s \times \quad BHP_t\sqrt{\frac{T_s}{T_t}}\right)$

The remainder of the correction to standard day rate of climb follows the jet check climb procedure. The fuel flow correction is not applied, and the final plot produced appears as Figure 2.14.

CLIMB PERFORMANCE SUMMARY
AIRCRAFT S/N
ENGINE SERIES – S/N
CLIMB SCHEDULE IDENTIFICATION
POUNDS OF FUEL ALLOTTED FOR TAXI, TAKEOFF
ACCELERATION TO CLIMB SPEED

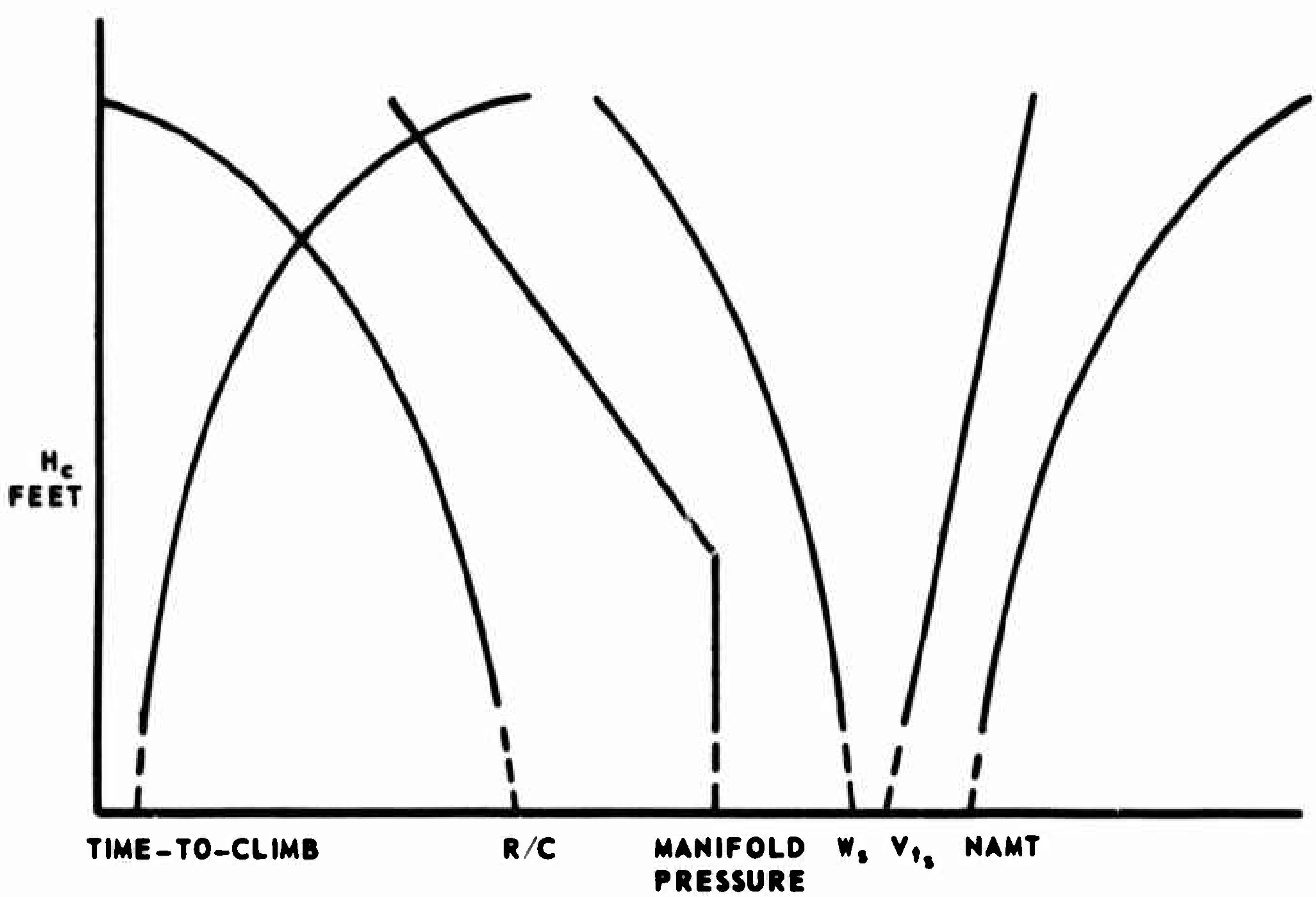


FIGURE 2.14

CHAPTER III

CRUISE PERFORMANCE TESTS

This section describes flight test techniques and data reduction methods for determining velocity, altitude, and power setting under cruise conditions. The tests discussed include a constant altitude method normally used to determine cruise data for a reciprocating engine aircraft and a constant weight - altitude ratio (W/δ) method normally used for jet engine powered aircraft cruise tests.

Test techniques described in this section pertaining to the constant (W/δ) speed - power test (the usual name applied to the cruise test) are written primarily for the single spool compressor - constant geometry engine, however, they apply equally as well to twin spool compressor - variable geometry engines. Data reduction outlines, on the other hand, apply only to fixed geometry engines. Two outlines are presented. One method utilizes engine speed as the power parameter which applies to the single spool compressor engine and the other utilizes the engine pressure ratio, P_{t_7}/P_{t_2}, as the power parameter. The latter method is primarily for twin spool compressor engines.

For more complex engines, the outlines described herein should be modified as necessary. Because of the variety of configurations that exist, it is not practical, nor possible, to describe methods for correcting engine data to standard conditions which are suitable for all types. Frequently, it is not immediately evident as to when dimensional analysis methods are applicable. The characteristics of each of the more complex engines should be studied so that methods may be modified for the individual case. Engine manufacturer's charts are a good source of data when making this analysis.

3.1 SPEED POWER - CONSTANT ALTITUDE METHOD ALTITUDE METHOD

The purpose of the speed power test, constant altitude method, is to determine the standard day level flight performance of reciprocating engine aircraft. Specifically, the following items are desired:

(1) Brake horsepower at full throttle operation throughout the altitude range of the aircraft.

(2) The maximum level flight true airspeed of the aircraft throughout its altitude range.

(3) The maximum manifold pressure available at different altitudes.

(4) The power required versus velocity obtainable at all weights and altitudes.

(5) The drag polar of the aircraft.

In general, this test consists of determining points on the power required versus true airspeed curve of the aircraft. The aircraft is stabilized at different speeds throughout its speed range while holding a constant altitude and sufficient data is recorded to determine the power output of the engine and other data listed above.

Because of the nature of the reciprocating engine - propeller combination, airplanes equipped with this means of propulsion are, for the most part, designed to fly at speeds less than drag divergence. Although Mach number

effects are present and do modify the lift slope at velocities less than that for drag divergence, there is no major modification in the lift-drag relationship until drag divergence is encountered. Thus, for all the important phases of flight testing of reciprocating engine aircraft the following basic assumption or relationship is valid:

$$C_D = C_{D_p} + \frac{C_L^2}{\pi ARe} = C_{D_p} + C_{D_i} \qquad \text{Eq 3.1}$$

The most convenient method of presentation of power required data for reciprocating engine aircraft is to present all data from all altitudes on a common sea level - standard weight plot. To show how this comes about, consider flight at the same value of C_L at sea level under standard conditions, at standard weight and at some altitude where the density (ρ) and the weight differ from standard. At sea level the density is ρ_o and the weight is W_s, so by letting V_{SL} = sea level standard airspeed, then

$$V_{SL} = \sqrt{\frac{2\,W_s}{\rho_o\, C_L\, S}} \qquad \text{Eq 3.2}$$

and

$$THP_{r_{SL}} = \frac{C_D\, \rho\, V^3\, S}{1100} = \frac{1}{550}\sqrt{\frac{2\,W_s^{\,3}\, C_D^{\,2}}{\rho_o\, S\, C_L^{\,3}}}$$

Eq 3.3

According to the assumption in Equation 3.1, the same C_D value must occur at altitude as at sea level if flying at the same C_L, therefore,

$$V = \sqrt{\frac{2\,W_t}{\rho\, C_L\, S}}$$

and

$$THP_r = \frac{C_D\, \rho\, V^3\, S}{1100} = \frac{1}{550}\sqrt{\frac{2W_t^{\,3}\, C_D^{\,2}}{\rho\, S\, C_L^{\,3}}}$$

Eq 3.4

Now, for the ratios of horsepower and speed,

$$THP_{r_{SL}} = THP_r \sqrt{\frac{\rho}{\rho_o}} \sqrt{\left(\frac{W_s}{W_t}\right)^3}$$

$$= THP_r \sqrt{\sigma} \sqrt{\left(\frac{W_s}{W_t}\right)^3} = PIW$$

Eq 3.5

and

$$V_{SL} = V_t \sqrt{\sigma} \sqrt{\frac{W_s}{W_t}} = VIW \qquad \text{Eq 3.6}$$

Now the PIW vs VIW curve can be plotted.

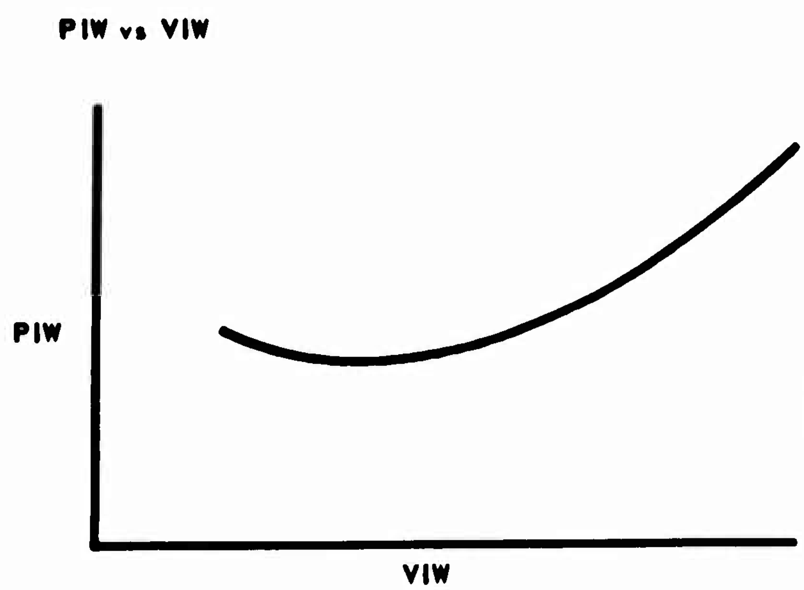


FIGURE 3.1

As in all flight test data reduction, test data has to be corrected to standard day conditions to be valid for comparison. The power setting for a given equivalent airspeed (V_e) differs with change of temperature for two reasons. One is that the THP_r for a given equivalent airspeed increases as the temperature increases. From aerodynamic theory it can be shown that the drag of an airplane at a given weight is constant for all altitudes and temperatures at a given V_e. From this, the change of THP_r from test day to standard day can be found.

By definition:

$$THP_r = \frac{D \times V_t}{550} \quad (V_t \text{ in ft/sec})$$

therefore:

$$THP_{r_s} = \frac{D_s \times V_{t_s}}{550} \qquad \text{Eq 3.7}$$

$$THP_{r_t} = \frac{D_t \times V_{t_t}}{550} \qquad \text{Eq 3.8}$$

Dividing Equation 3.7 by 3.8

$$\frac{THP_{r_s}}{THP_{r_t}} = \frac{V_{t_s}}{V_{t_t}}$$ (since $D_s = D_t$ at the same V_e and assuming $W_t = W_s$)

It can also be shown that at a constant pressure altitude for a given V_e:

$$V_{t_s}/V_{t_t} = \sqrt{\frac{T_s}{T_t}}$$

therefore the THP_r equation becomes

$$THP_{r_s} = THP_{r_t}\sqrt{\frac{T_s}{T_t}} \qquad \text{Eq 3.9}$$

The second temperature effect is the effect on the power output of an engine. The BHP_a of a reciprocating engine with standard day air temperature at the carburetor can be determined from the engine charts. This power depends on manifold pressure, rpm, and altitude. If the airplane is flown on a test day the BHP_a, which the engine delivers, will differ from the BHP_a obtained from the engine chart when at the same power settings and altitude. The following equation for the correction of BHP_a from chart values to test day conditions is,

$$BHP_{a_t} = BHP_{a_s}\sqrt{\frac{T_s}{CAT_t}} \qquad \text{Eq 3.10}$$

where BHP_s is BHP obtained from engine charts.

In the data reduction all instrument readings are corrected for instrument error and applicable position error before determining the BHP that the engine was delivering for the test point. This BHP is found by reading the BHP from the engine chart and correcting this value with the use of Equation 3.10.

In straight and level stabilized flight the THP_a is equal to the THP_r (since drag is equal to thrust). The assumption is made that propeller efficiency (η_p) does not change from test to standard day temperature at the same power setting and V_e; therefore $BHP_r = THP_r/\eta_p$. Thus $BHP_a = BHP_r$ for the test point, and BHP_a can be determined as shown above.

Since $THP_{r_s} = THP_r\sqrt{T_s/T_t}$ from Equation 3.9 the BHP_{r_s} can be calculated from values of BHP_{r_t}. Then a plot can be constructed of BHP_{r_s} vs V_e which is the power required curve for standard conditions. It was obtained by using the data on a test day. This correction method ignores the effect of weight on the power required curve but is useful for airplanes which do not change weight an appreciable amount during the flight test. Later in the data reduction, a PIW - VIW curve will be obtained which considers the effect of weight.

PRE-FLIGHT PREPARATION:

Now that is has been determined what is required in the way of data, the techniques which should be used to obtain good data will be discussed. Before the flight, the pilot should ensure that he knows exactiy what data are going to be obtained and under what conditions he is to obtain it. The pilot will note that efficient planning will cut down the flight time required to obtain data, therefore, the most efficient use of aircraft time is obtained.

The speed power test should be run at four altitudes ranging from near sea level to near the maximum altitude of the aircraft. Plans should be made to

obtain at least five stabilized points at each altitude. At least two of these points should be in the low speed range or the back side of the power required curve. One maximum speed point should be run at altitudes half way between the test altitudes.

The items to be recorded should be noted and a convenient flight data card prepared to allow rapid data recording. The following data card format is suggested:

1. FUEL COUNTER READING BEFORE ENGINE START_______.

2. CRITICAL ALTITUDE

H_i______; V_i______; FAT_______.

3. BEFORE TEST AT V_{min} AND TEST ALTITUDE READ: FAT____.

4. RUN NO. 1 2 3 4 5 ODD ALT.
RPM
MP
CAT
FAT
FUEL
H_i
V_i
TOD

5. AFTER TEST AT V_{min} AND TEST ALTITUDE READ: FAT____.

Items 3 and 5 of the preceding data card are for the purpose of prorating any change in the ambient temperature for a temperature probe calibration.

Plans should be made to fly the test in the same relative air mass, i.e., the same general area over the ground and away from mountains. Smooth air is also essential for the flying of this test.

On the reverse side of the data card, a set of axis should be drawn where manifold pressure and indicated airspeed can be plotted. Such a plot is valuable in that consistency of data can be determined while in flight. If at any point the curve is not uniform or smooth the point can be reflown. This procedure will prevent the necessity of rerunning the complete flight.

IN-FLIGHT TECHNIQES:

Following are the recommended procedures and techniques for determining the speed versus power using the constant altitude method of testing:

(1) Before engine start record the fuel counter reading.

(2) After takeoff, trim aircraft to zero stick forces at best climb speed.

(3) Record data required at critical altitude.

(4) On arriving at the test altitude the aircraft should be stabilized at near V_{min} so that the FAT can be recorded. Since the aircraft is already stabilized at this point it is possible to record the necessary data listed in part 4 of the suggested data card.

(5) The aircraft can be slowed down and a minimum speed point can be obtained before proceeding to the maximum speed points.

(6) Accelerate the aircraft at full throttle and obtain a maximum speed point. Record the necessary data.

(7) Reduce manifold pressure in 3 to 4 inch increments letting airspeed stabilize each time before recording the necessary data. Repeat this process to obtain the necessary points. After each stabilized point, the manifold pressure is plotted versus indicated airspeed on the reverse side of the data card to check consistency.

(8) After completion of the required runs at test altitude repeat the first run exactly to obtain the FAT.

(9) Make one high speed run 2,500 feet below the test altitude and record the required data.

DATA REDUCTION OUTLINE:

Determine static free air temperature at the test altitude and, from previously discussed procedures, determine equivalent airspeed. Then determine corrected engine RPM, manifold pressure, and carburetor air temperature readings.

Column	Symbol	Units	Description
(1)	BHP_c		BHP_s by using engine chart
(2)	T_s	°K	Standard temperature at test altitude.
(3)	BHP_t		Test BHP is equal to $BHP_s \times \sqrt{\frac{T_s}{CAT_{ic}}}$
(4)	BHP_{Plot}		$BHP_t \times \sqrt{\frac{T_s}{T_{a_t}}}$

Plot V_e vs (4). Draw a curve for each test altitude plus the high speed run half way between the test altitudes (indicated by the dashed lines).

V_e vs BHP_{plot}

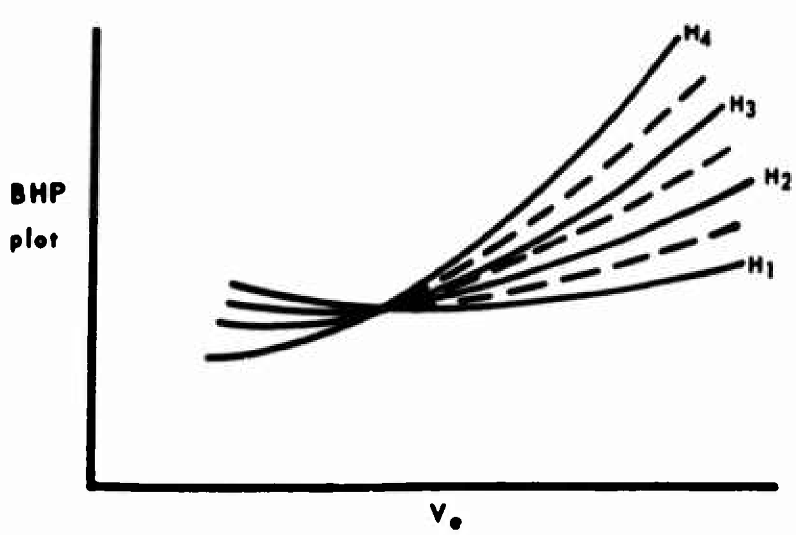


FIGURE 3.2

Column	Symbol	Units	Description
(5)	$(1/\sqrt{\sigma})_t$		
(6)	W_s	lb	Standard gross weight (standard takeoff weight).
(7)	W_f	lb	Fuel used
(8)	W_t	lb	$W_s - W_f$
(9)	$(W_t/W_s)^{1/2}$		
(10)	$(W_t/W_s)^{3/2}$		
(11)	VIW	kt	$V_e \Big/ \sqrt{\frac{W_t}{W_s}}$
(12)	PIW		$\frac{BHP_t \, \eta_p \sqrt{\sigma}}{\sqrt{(W_t/W_s)^3}}$

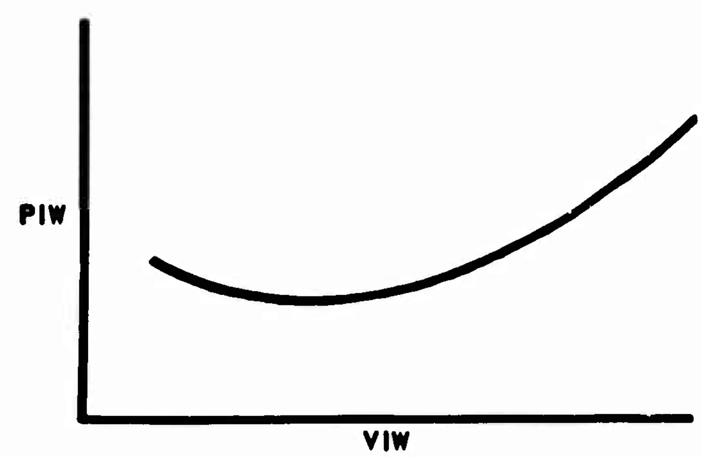


FIGURE 3.3

The drag polar and the Oswald efficiency factor (e) can now be found for the wing from data contained in the PIW - VIW plot.

Column	Symbol	Units	Description
(13)	VIW	kt	Arbitrary values of VIW from Figure 3.3
(14)	PIW		Values of PIW corresponding to VIW in (13)
(15)	VIW^2	ft/sec	$(ft/sec)^2$
(16)	VIW^3	ft/sec	$(ft/sec)^3$

$$(17)\quad C_D \qquad \frac{2 \times 550 \times PIW}{\rho_o\, S\, (VIW)^3}$$

$$(18)\quad C_L \qquad \frac{2\, W_s}{\rho_o\, (VIW)^2\, S}$$

$$(19)\quad C_L^{\,2}$$

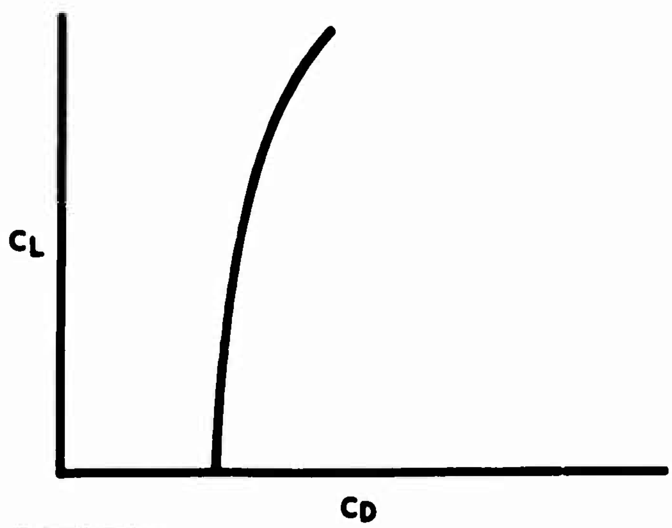


FIGURE 3.4

If C_D is plotted vs $C_L^{\,2}$, the equation becomes a straight line with C_{D_p} the intercept on the C_D axis where $C_L^{\,2} = 0$. The slope of the line is $1/AR\ e$; therefore, knowing the aspect ratio, Oswald's efficiency factor may be determined. $e = 1/\pi A.R.\ \text{slope}$, where $A.R. = b^2/S$.

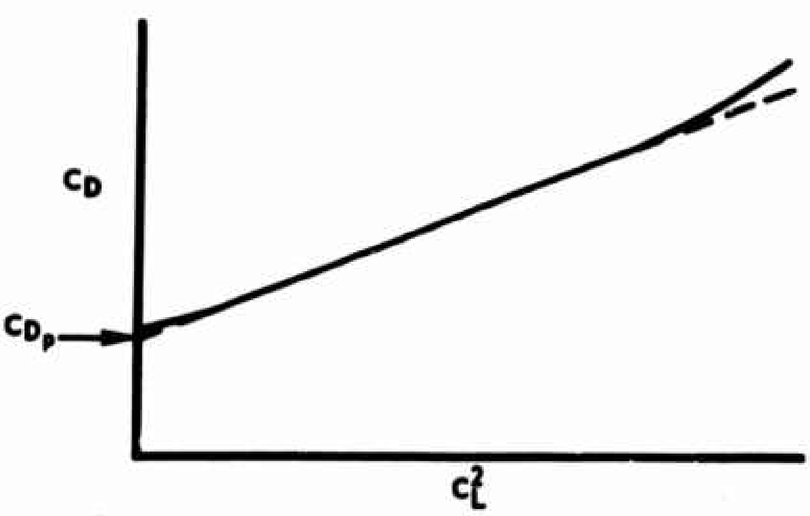


FIGURE 3.5

In actual practice the plot of C_D vs $C_L^{\,2}$ will only be a straight line during the middle portion. The breakoffs at the low C_L, (high speed end), is due to a decrease in propeller efficiency, (η_p), and the effect of drag divergence. At the high C_L, (low speed end), the breakoff is due to a decrease in propeller efficiency and an increase in C_{D_p}. The "straight line" theory assumes a constant η_p, e, and C_{D_p} and speeds below the drag rise.

3.2 SPEED POWER AT CONSTANT W/δ

The constant W/δ method of testing is used to determine the standard day level flight performance of the turbo-jet aircraft. In particular the following specific performance characteristics are tested for:

(1) The fuel flow characteristics for all weights, speeds, and altitude conditions.

(2) Drag characteristics for all weights, speeds, and altitude conditions.

(3) The specific range characteristics for all weights, speeds, and altitude conditions.

The aircraft drag parameter is a function of Mach number and the weight parameter as shown:

From aerodynamics:

$$\frac{D}{P_a} = f\left(\frac{W}{P_a}, M\right)$$

From the Buckingham π analysis for jet engines (single spool):

$$\frac{F}{P_a} = f\left(\frac{N}{\sqrt{T_a}}, M\right)$$

Since $\qquad D = F_n$

Then

$$f\left(\frac{W}{P_a}, M\right) = f\left(\frac{N}{\sqrt{T_a}}, M\right)$$

$$\frac{N}{\sqrt{T_a}} = f\left(\frac{W}{P_a}, M\right)$$

or

$$\frac{N}{\sqrt{\theta}} = f\left(\frac{W}{\delta}, M\right)$$

The test program will cover the range of airspeeds for specific values of $N/\sqrt{\theta}$ and W/δ, and will present the relationship between these parameters and Mach number. This will give the relationship between true airspeed, engine speed, and altitude at standard weight and temperature.

In general, the test consists of stabilizing at different airspeeds and power settings while maintaining a constant W/δ. To obtain points near the minimum drag airspeed there may be some difficulty in stabilizing on the airspeed; therefore, the approximate airspeed and engine speed should be known. These circumstances are more likely to occur at high altitud s where the speed for maximum endurance and even maximum range may be at the minimum drag point.

At low speeds, below the speed for minimum drag, a jet aircraft will not stabilize well. This is brought about because drag increases as speed decreases and a condition may occur where at a constant engine speed the airspeed may slowly decrease until a stall is reached. The aircraft cannot be successfully operated in this range, but flights are required in order to plot points on the back side of the drag curve. It is acceptable to allow a slight descent or climb (less than 200 ft/min) to maintain airspeed. This method usually gives more reliable data because hysteresis or elastic lag effects in the altimeter are almost eliminated.

For the best results the W/δ should be maintained as close as possible to the desired value, however $\pm$ 2 % is entirely satisfactory.

PRE-FLIGHT PREPARATION:

In order to fly the test at a constant W/δ certain preflight preparations must be made. It is necessary for the pilot to have available to him certain charts relating fuel counter to altimeter reading for a constant W/δ. Consideration should also be given to both altitude position and instrument error. In addition, care should be given in preparing a suitable flight data card. This test is well suited for hand recorded data.

The following data will be required before the necessary charts and tables are prepared:

(1) The empty weight of the aircraft.

(2) Fuel density and fuel loading.

(3) Altimeter calibrations relevant to altitude and airspeed of the test.

(4) Airspeed calibrations.

The following procedures may be used to obtain the charts required to perform the test:

(1) Determine the standard altitude (H_{c_s}) at which the test is to be flown and obtain the corresponding δ from standard atmospheric tables.

(2) Determine the standard gross weight (W_s) for the test. This is the average weight from the start of the test until the end of the test.

(3) Calculate $(W/\delta)_s$.

(4) Obtain the values for the following table:

Altitude	Pressure Ratio	Gross Weight	W/δ
H_{c_s} + 2000'	δ_1	W_1	$(W/\delta)_s$
H_{c_s} + 1000'	δ_2	W_2	$(W/\delta)_s$
H_{c_s}	δ_s	W_s	$(W/\delta)_s$
H_{c_s} - 1000'	δ_3	W_3	$(W/\delta)_s$
H_{c_s} - 2000'	δ_4	W_4	$(W/\delta)_s$

Example:

$$W_1 = (W/\delta)_s \times \delta_1$$

(5) Construct a plot of H_c vs Gross Weight. This plot can be used to determine altitude to fly for W/δ = Constant at any value of aircraft gross weight.

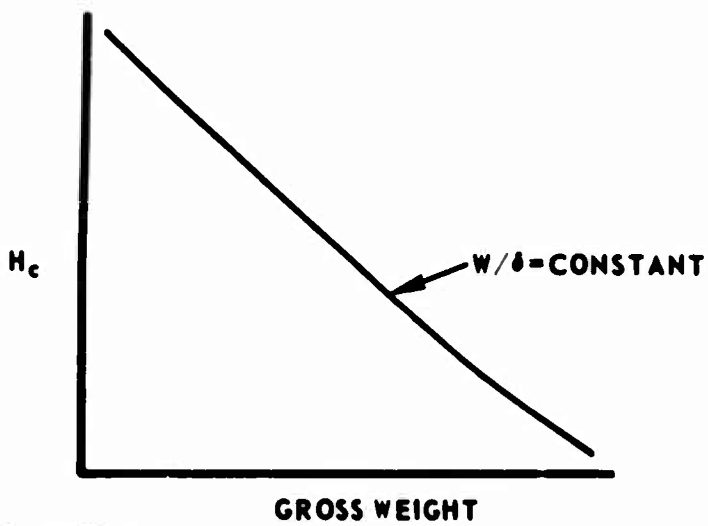


FIGURE 3.6

(6) Using the above graph, plot F/C readings vs Gross Weight on the right hand ordinate. If fuel temperature changes throughout the flight, use an average value for determining fuel density.

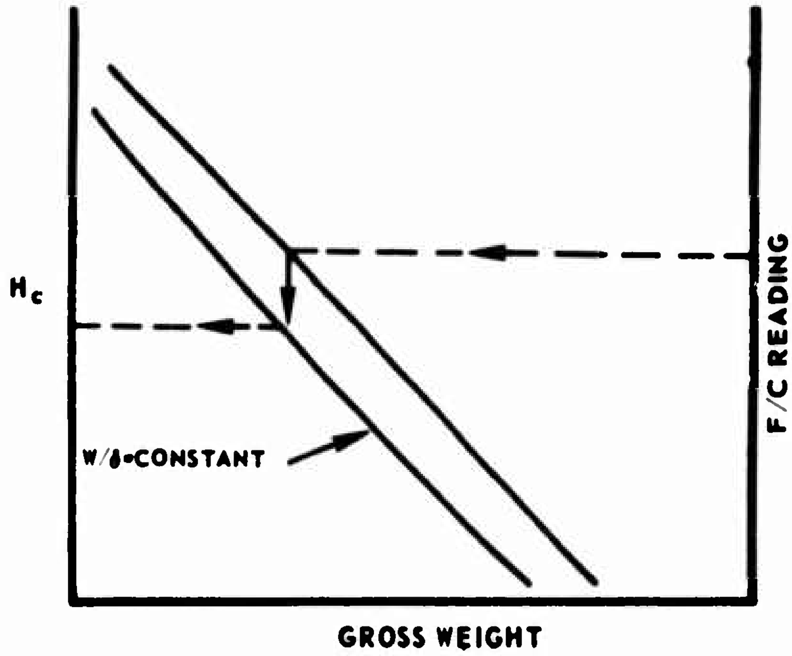


FIGURE 3.7

This chart will yield the correct altitude to fly at any value of fuel counter reading, for a given W/δ.

Note that the curve of fuel counter reading vs Gross Weight is a straight line and is dependent upon the basic weight of the aircraft. If the basic weight of the aircraft changes or the test is flown in another aircraft, this curve (F/C reading vs gross weight) can be easily changed.

(7) Since the values read from the above chart are "altitudes to fly," curves of ΔH_{pc} vs V_{ic} and ΔH_{ic} vs H_i should be utilized. For convenience, these two curves can be resolved into one (for the altitudes and airspeeds concerned) and plotted on the chart of H_c vs gross weight vs F/C. Particular attention should be given to the sign (sense) of this correction because the above procedure necessitates going from calibrated values to indicated values.

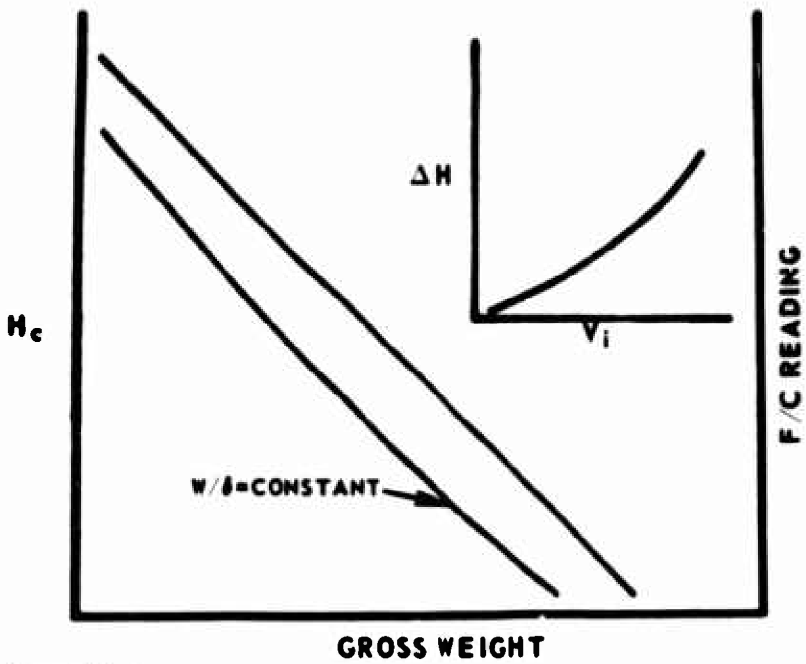


FIGURE 3.8

(8) Below is a suggested data card to be used by the pilot, showing typical entries:

AIM V_i KTS	ACTUAL V_i KTS	ALT/Ft	F/C #1 (START OF RUN)	F/C #2 (END OF RUN)	SEC T	FAT °C	N_i %
V_{max}	456.5	18540	3412	3306	1:06.2	20.0	99.9
4.0	411.0	18810	3217	3100	1:14.5	16.5	96.4
370	368.5	19060	3020	2902	1:25.3	12.0	93.7

IN-FLIGHT TECHNIQUES:

The following is a recommended procedure for performing the speed power test using the constant W/δ method so that flight time may be used efficiently:

(1) Before engine start the pilot should assure himself of the correct fuel loading and that the fuel counter is set correctly.

(2) When approaching within 2000 to 3000 feet of the test altitude, read the fuel counters and obtain an "altitude to fly" for the first data point. Allow 20 to 30 gallons or 100 - 200 pounds for the airspeed to stabilize.

(3) The correction for ΔH_{pc} and ΔH_{ic} should be read from the appropriate chart using the aim airspeed and altitude. Example: If 19,855 feet corresponds to 3100 pounds of fuel the pilot should be at 19,855 feet with at least 3200 pounds of fuel. This procedure would allow the airspeed to stabilize and also the pilot will be able to record the fuel flow by the time the correct W/δ is reached.

(4) To obtain the fuel flow, re-

cord the fuel counter reading (F/C # 1) and start the stop watch as the aircraft approaches the stabilized airspeed (± 3 knots). Fly the aircraft at the required altitude until the fuel counter reading (F/C # 2) comes up for the required W/δ (3100 pounds in the example above). Record the time between F/C #1 and F/C #2. A minimum fuel increment of between 6 and 10 gal or 40 and 60 lb should be used. Record all other necessary data. NOTE: The pilot should be absolutely certain that the aircraft is stabilized before recording data.

(5) Obtain enough stabilized points to define a power vs velocity curve that covers the entire range of the aircraft at the particular altitude tested.

(6) The pilot can facilitate and expedite the stabilizing of the aircraft by proper trimming, pitch control by outside reference, and recording data in an organized sequence. The aircraft should be trimmed for hands off flight when stabilized. Altitude control on the front side points and airspeed control on back side points can be controlled precisely by the attitude method. It will be found that the majority of the data may be recorded while waiting for the aircraft to stabilize.

(7) To assure the pilot that the test data he has taken will give even increments of points he should plot on the back side of a data card N_i vs V_i. This will give an early reference as to the shape of the power required curve.

■ DATA REDUCTION OUTLINE UTILIZING ENGINE SPEED AS THE POWER PARAMETER:

Determine calibrated airspeeds, calibrated altitudes, Mach numbers at the test altitudes and airspeeds, true airspeeds at the standard altitude, instrument corrected temperatures and ambient temperatures at each test point. NOTE: The following outline assumes that the test was flown within 2% of the standard W/δ. No corrections are presented for ΔW/δ.

Column	Symbol	Units	Description
1	$(\sqrt{\theta})_t$	--	$\sqrt{\dfrac{(T_a)_t}{288}}$
2	W_t	lb	Original data at each test point.
3	$(\delta)_t$	--	$\dfrac{P_a}{29.92}_t$
4	$(W/\delta)_t$	--	--
5	N_{ic}	RPM	Original data corrected for instrument error
6	$\left(\dfrac{N}{\sqrt{\theta}}\right)_t$	--	$\left(\dfrac{N_{ic}}{\sqrt{\theta}}\right)_t$

Plot Mach number vs $\left(\dfrac{N}{\sqrt{\theta}}\right)_t$ at constant W/δ

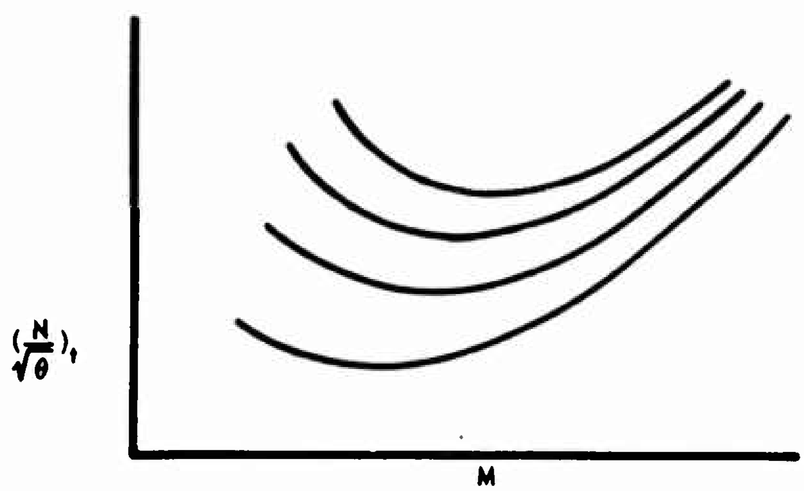


FIGURE 3.9

7	$(H_c)_s$	ft	Altitude for the standard W/δ
8	$(V_c)_s$	kt	Use test M and $(H_c)_s$ at standard W/δ
9	$(\sqrt{\theta})_s$		$\sqrt{\frac{(T_a)_s}{288}}$ - from standard T at H_{c_s}
10	N_s	RPM	$\left(\frac{N_{ic}}{\sqrt{\theta}}\right)_t \times (\sqrt{\theta})_s$
11	$(\delta)_s$		$\frac{P_a}{29.92}$ at $(H_c)_s$
12	$(W_t)_s$	pounds	Standard weight at which the constant W/δ was computed
13	$(W/\delta)_s$	--	--

Plot N_s vs $(V_c)_s$ at constant $(W/\delta)_s$

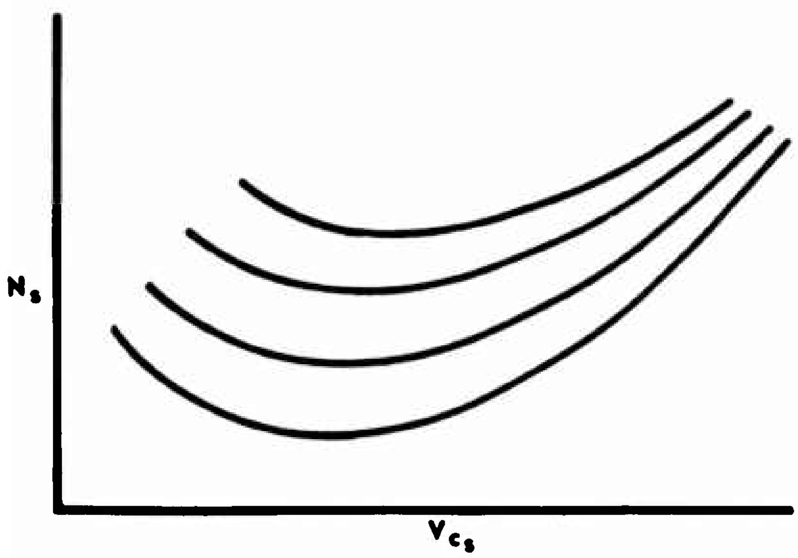


FIGURE 3.10

14	$(W_f)_t$	lb/hr	Fuel flow in pounds per hour at each test point
15	$(T_{t2})_t$	°K	Total temperature at the face of the compressor. Assume $(T_{t2})_t = T_{ic}$
16	$\sqrt{(\theta_{t2})_t}$	--	$\sqrt{\left(\frac{T_{t2}}{288}\right)_t}$
17	$(T_{t2})_s$	°K	$\dfrac{(T_a)_s \text{ or } T_{ic}}{T_a/T_{ic}}$ at standard altitude
18	$\sqrt{(\theta_{t2})}_s$	--	$\sqrt{\left(\frac{T_{t2}}{288}\right)_s}$
19	$(P_{t2}/P_a)_t$	--	$P_o = P_{t2}$ = compressor inlet pressure. Obtain from chart of M vs P_o/P_a at test Mach numbers and % ram efficiency
20	$(P_{t2})_t$	in Hg	$(P_a)_t \times \left(\frac{P_{t2}}{P_a}\right)_t$
21	$(\delta_{t2})_t$	--	$\frac{(P_{t2})_t}{29.92}$
22	$(P_{t2})_s$	in Hg	Total pressure at the compressor face at the standard altitude
23	$(\delta_{t2})_s$	--	$\frac{(P_{t2})_s}{29.92}$
24	$(W_f)_s$	lbs/hr	$\dfrac{(W_f)_t \times (\delta_{t2})_s \sqrt{(\theta_{t2})_s}}{(\delta_{t2})_t \sqrt{(\theta_{t2})_t}}$
25	Nam/lb	--	$\frac{(V_t)_s}{(W_f)_s}$ Specific range - nautical air miles per pound of fuel

Plot NAM/Lb vs $(V_t)_s$ at constant $(W/\delta)_s$.

On the same plot put lines of constant RPM, speed for maximum range and speed for recommended cruise. (The recommended cruise speed is approximately 3 to 5% higher than the speed for maximum range or at a V_t which gives 99% of best specific range.)

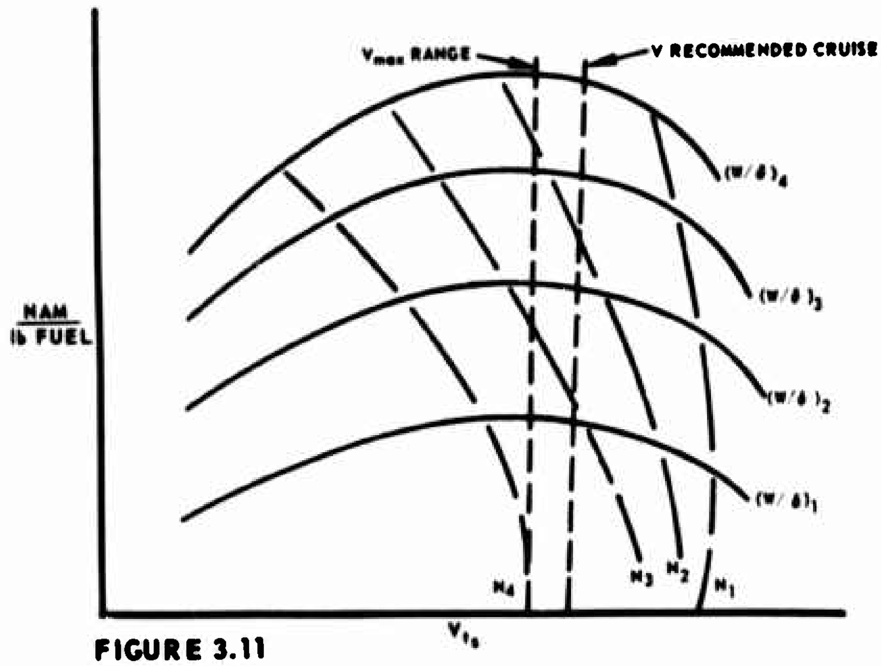


FIGURE 3.11

DATA REDUCTION OUTLINE UTILIZING ENGINE PRESSURE RATIO AS THE POWER PARAMETER:

Data reduction presented in this section is used primarily for twin spool compressor engines, however it is valid for any engine if pressure ratio instrumentation is available. Also the correction for $\Delta W/\delta$ is valid only for fixed geometry engines. Engine manufacturer's data should be consulted to determine methods of correction to standard day for any given variable geometry engine.

In a dual compressor engine, the low pressure rpm (N_l) controls the airflow through the engine, but is not mechanically connected to the high pressure compressor. The high pressure compressor rpm (N_2) is controlled by the throttle and the low pressure compressor "floats". The ratio of rpm between N_1 and N_2 is very likely to be different from one set of conditions (or from one engine) to the next, therefore, the parameter $N/\sqrt{\theta}$ is not valid and is replaced by P_{t_7}/P_{t_2}. P_{t_7}, as used in this data reduction outline, is the turbine exhaust pressure and P_{t2} is the low pressure compressor inlet pressure.

DATA REDUCTION OUTLINE:

Determine V_c, H_c, M, T_{ic}, T_{a_t} and T_{a_s}.

Column	Symbol	Units	Description
1	T_{t2_t}	^oK	$T_{a_t}(1 + .2M^2)$ assume K = 1.0
2	T_{t2_s}	^oK	$T_{a_s}(1 + .2M^2)$ assume K = 1.0
3	$\sqrt{\theta_{a_s}}$		$\sqrt{\frac{T_{a_s}}{288}}$
4	$\sqrt{\theta_{a_t}}$		$\sqrt{\frac{T_{a_t}}{288}}$
5	$\sqrt{\theta_{t2_s}}$		$\sqrt{\frac{T_{t2_s}}{288}}$
6	$\sqrt{\theta_{t2_t}}$		$\sqrt{\frac{T_{t2_t}}{288}}$
7	$N_{l_{ic}}$	RPM	Low Pressure compressor RPM instrumented corr.
8	$N_{l_{ic}}/\sqrt{\theta_{t2_t}}$		
9	$\left(\frac{W_a\sqrt{\theta_{t2}}}{\delta_{t2}}\right)_t$	#/sec	From air flow curves using the low pressure rotor rpm parameter

AIR FLOW CURVES

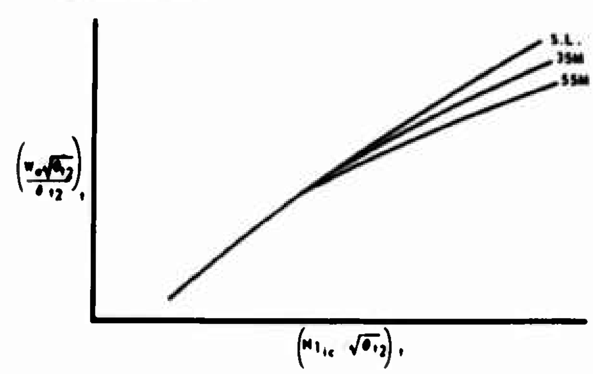


FIGURE 3.12

10	W_a	#/sec	$W_a = \left(\frac{W_a \sqrt{\theta_{t_2}}}{\delta_{t_2}}\right)_t \times \left(\frac{\delta_{t_{2_t}}}{\sqrt{\theta_{t_{2_t}}}}\right)$
11	$\left(P_{t_2}/P_o\right)_t$		From inlet duct recovery curves using Mach number and the airflow parameter.

INLET DUCT RECOVERY CURVES

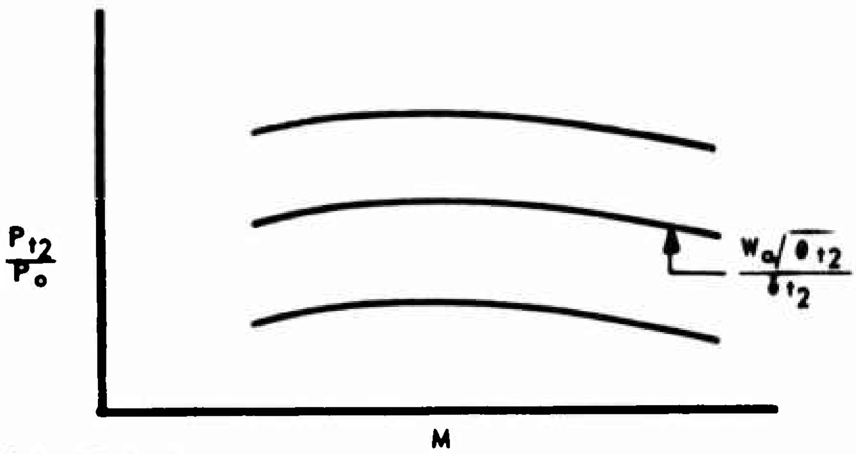


FIGURE 3.13

12	$\left(P_{t_2}/P_a\right)_t$		$\left(P_{t_2}/P_{t_0}\right)_t \left[\left(1 + .2M^2\right)^{3.5} - 1\right] + 1$
13	P_{a_t}	in. Hg	From Charts
14	$P_{t_{2_t}}$	in. Hg.	$\left(P_{t_2}/P_a\right)_t \times \cdot P_a$
15	W_t	pounds	Original data
16	W_s	pounds	Standard weight at which W/δ was computed
17	$(H_c)_s$	feet	Altitude for standard W/δ
18	δ_{a_t}		$\frac{P_{a_t}}{29.92}$
19	δ_{a_s}		Use Standard atmosphere tables
20	$\delta_{t_{2_t}}$		$\frac{P_{t_{2_t}}}{29.92}$
21	$\delta_{t_{2_s}}$		$\delta_{a_s} \times \frac{1}{\left(P_{t_2}/P_a\right)_t}$
22	V_{t_t}	knots	$38.967\, M \sqrt{T_{a_t}}$
23	V_{t_s}	knots	$38.967\, M \sqrt{T_{a_s}}$
24	$W_{f_{t_{eng}}}$	#/hr	From original data
25	$W_{f_{t_{A.B.}}}$	#/hr	From original data
26	$W_{f_t}/\delta_{t_{2_t}}\sqrt{\theta_{t_{2_t}}}$ ENG		
27	$W_{f_t}/\delta_{t_{2_t}}\sqrt{\theta_{t_{2_t}}}$ A. B.		
28	$\left(P_{t_7}\right)_t$	in. Hg.	From original data
29	$\left(P_{t_7}/P_{t_2}\right)_t$		

Plot $\left(W_f/\delta_{t_2}\sqrt{\theta_{t_2}}\right)_t$ vs $\left(P_{t_7}/P_{t_2}\right)_t$ for both afterburner and engine

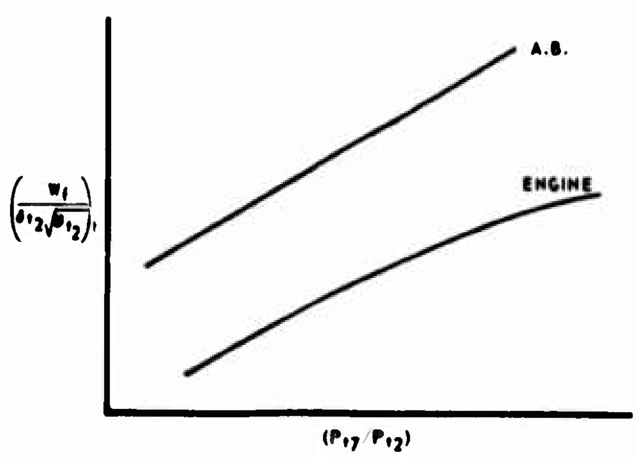


FIGURE 3.14

30 $\left(W/\delta_a\right)_s$

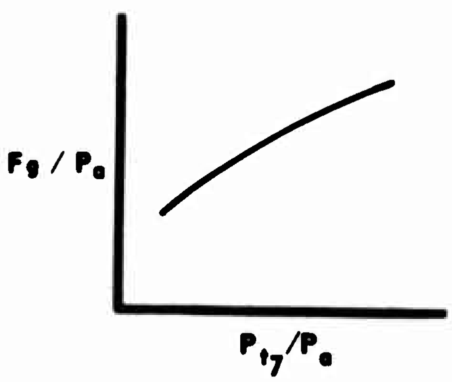
31 $\left(W/\delta_a\right)_t$

32 $\left(\Delta W/\delta_a\right)$ $\left(W/\delta_a\right)_s - \left(W/\delta_a\right)_t$

33 $\left(P_{t_7}/P_a\right)_t$

34 F_{e_t} pounds $.0525\ V_{t_t}\ W_a$ Reference Chapter 3 of AFTR-6273

35 $\left(F_g/P_a\right)_t$ Go into pressure probe calibration (obtained from static thrust run) at $\left(P_{t_7}/P_a\right)_t$



36 F_{g_t} pounds $\left(\frac{F_g}{P_a}\right)_t \times P_{a_t}$

37 F_{n_t} pounds $F_{g_t} - F_{e_t}$

38 $\left(F_n/\delta_a\right)_t$ $F_{n_t} \times 1/\delta_{a_t}$

The following steps are used to correct the test day W/δ and F_n to standard day W/δ and F_n.

39 C_{L_s} $\left(W/\delta_a\right)_s \left(\frac{.000675}{M^2\ S}\right)$

40 C_{L_t} $\left(W/\delta_a\right)_t \left(\frac{.000675}{M^2\ S}\right)$

41 C_{D_t} $\left(\frac{F_n}{\delta_a}\right)_t \left(\frac{.000675}{M^2\ S}\right)$

42 ΔC_L^2 $C_{L_s}^2 - C_{L_t}^2$

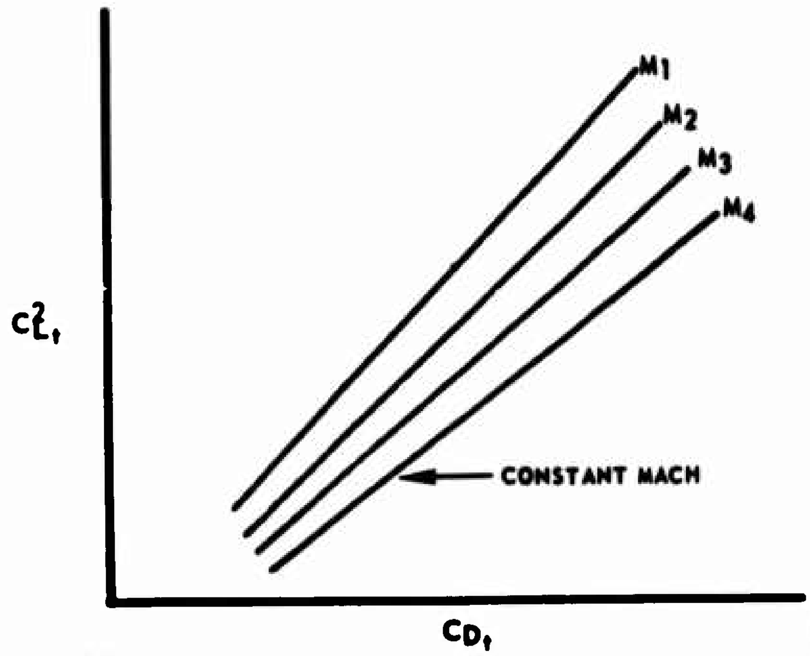
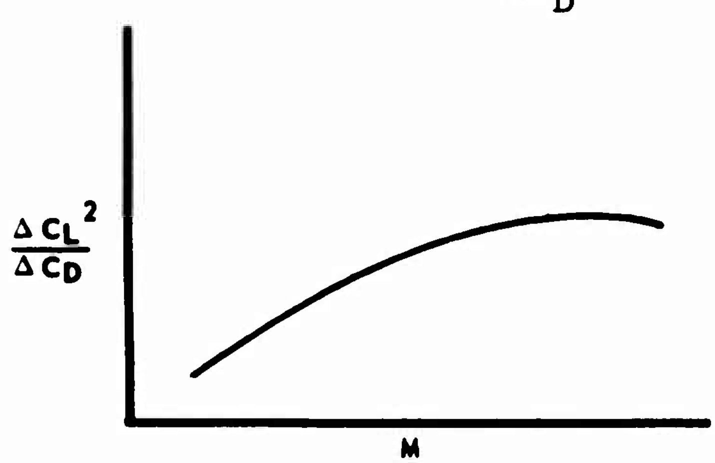


FIGURE 3.15

43 ΔC_D $\Delta C_L^2 \div \frac{\Delta C_L^2}{\Delta C_D}$ obtain $\frac{\Delta C_L^2}{\Delta C_D}$ from linear-ized drag polar at Mach (Figure 3.15). Make a working plot of $\frac{\Delta C_L^2}{\Delta C_D}$ vs M.



44	$\Delta(F_n/\delta_a)$	pounds	$\Delta C_D \times \dfrac{M^2\ S}{.000675}$
45	$(F_{n_s}/\delta_a)_s$	pounds	$\left(\dfrac{F_{n_t}}{\delta_a}\right)_t + \Delta\left(\dfrac{F_n}{\delta_a}\right)_t$
46	F_{n_s}	pounds	$\left(\dfrac{F_{n_s}}{\delta_a}\right)_s \times (\delta_a)_s$
47	$\Delta(F_n/P_a)$		Assume $\Delta(F_n/P_a) = \Delta(F_g/P_a) = \Delta\left(\dfrac{F_n}{\delta_a}\right) \div P_o$
48	$(F_g/P_a)_s$		$(F_g/P_a)_t + \Delta(F_n/P_a)$
49	$(P_{t_7}/P_a)_s$		Go into pressure probe calibration at $(F_g/P_a)_s$

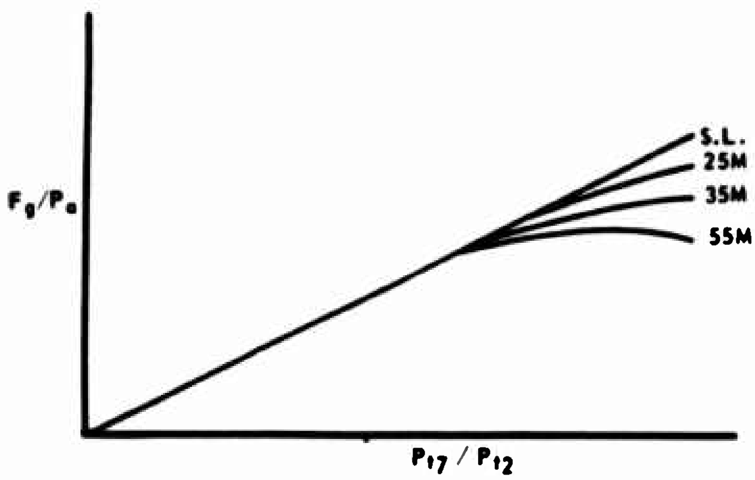


FIGURE 3.16

50	$\Delta(P_{t_7}/P_a)$	$(P_{t_7}/P_a)_s - (P_{t_7}/P_a)_t$
51	$\Delta(P_{t_7}/P_{t_2})$	$\Delta(P_{t_7}/P_a) \times (P_a/P_{t_2})$
52	$(P_{t_7}/P_{t_2})_s$	$(P_{t_7}/P_{t_2})_t + \Delta(P_{t_7}/P_{t_2})$

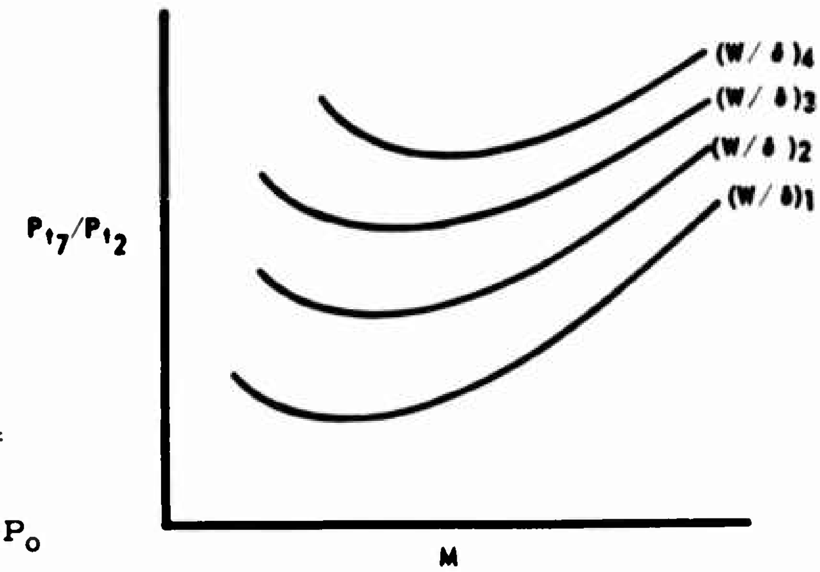


FIGURE 3.17

53	(P_{t_7}/P_{t_2})	Plot	Arbitrary values of P_{t_7}/P_{t_2} from Figure 3.17
54	$\dfrac{\Delta\left(W_f/\delta_{t_2}\sqrt{\theta_{t_2}}\right)}{\Delta(P_{t_7}/P_{t_2})}$		Slopes of Figure 3.17 at arbitrary values of P_{t_7}/P_{t_2}

MAKE A PLOT OF:

$$\left[\frac{\Delta\left(W_f/\delta_{t_2}\sqrt{\theta_{t_2}}\right)}{\Delta(P_{t_7}/P_{t_2})}\right]_{ENG}$$

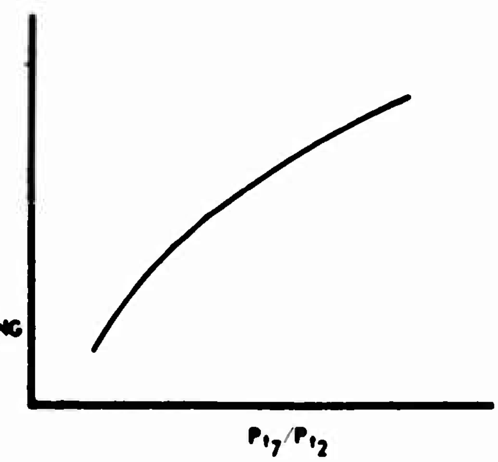


FIGURE 3.18

55	$\Delta\left(W_f/\delta_{t_2}\sqrt{\theta_{t_2}}\right)$	ENG	Go into Figure 3.18 at $(P_{t_7}/P_{t_2})_t$ and obtain $\dfrac{\Delta\left(W_f/\delta_{t_2}\sqrt{\theta_{t_2}}\right)}{\Delta(P_{t_7}/P_{t_2})}$ then multiply by $\Delta(P_{t_7}/P_{t_2})$

56	$\left(W_{f_s}/\delta_{t_2}\sqrt{\theta_{t_2}}\right)_{ENG}$		$\left(W_{f_t}/\delta_{2_t}\sqrt{\theta_{2_t}}\right)_{ENG} + \Delta\left(W_f/\delta_{t_2}\sqrt{\theta_{t_2}}\right)_{ENG}$
57	W_{f_s}	#/hr	$\dfrac{W_{f_t}}{\delta_{t_{2_t}}\sqrt{\theta_{t_{2_t}}}} \times \delta_{t_{2_s}}\sqrt{\theta_{t_{2_s}}}$
58	NAMPP		V_{t_s}/W_{f_s}

Plot NAMPP vs Mach at constant $(W/\delta)_s$. Point out Mach for recommended cruise.

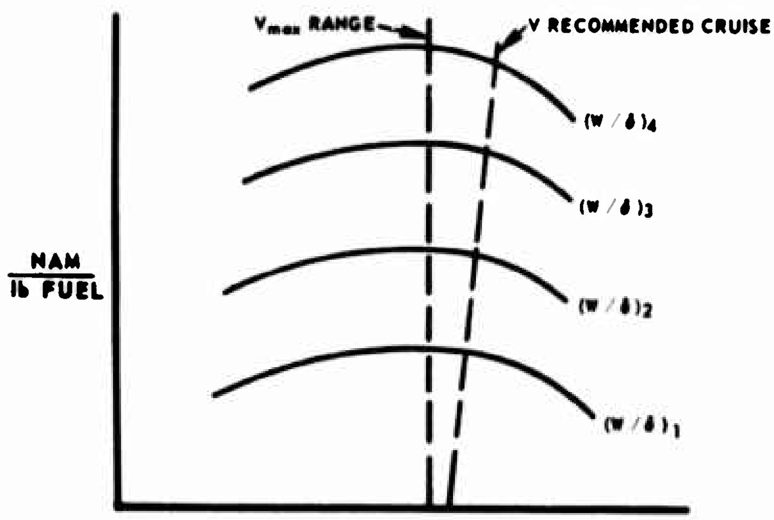


FIGURE 3.19

3.3 RANGE CRUISE CONTROL TEST FOR TURBOJET AIRCRAFT

The range cruise control flight test is used to determine the following range characteristics of a turbojet aircraft of a specific configuration.

(1) Check the W/δ where the optimum range factor occurs.

(2) The range available during the cruise portion of the test.

The actual range of a turbojet aircraft can be evaluated by examination of the specific range parameter, V_t/W_f, which has the units of nautical air miles per pound of fuel. Data obtained from the speed power constant W/δ test is shown below.

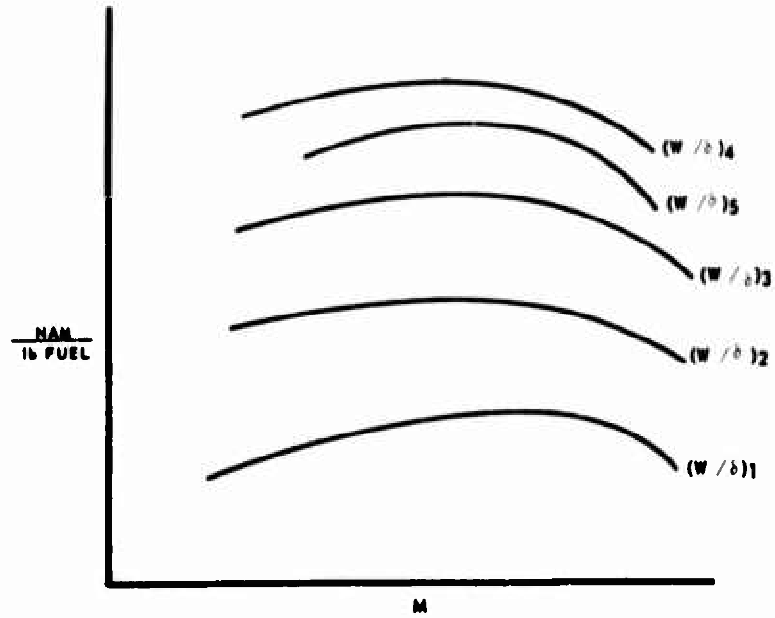


FIGURE 3.20

Thus it is seen that an increase in specific range is obtained with an increasing W/δ. However, a large W/δ can be reached where the specific range parameter will decrease, reference Figure 3.20, curve number 5.

The range factor of an aircraft is defined as RF - specific range x aircraft gross weight.

$$RF = \frac{V_t}{W_f} \times W_t$$

From theory, range at a constant W/δ is determined by,

$$\text{Range} = \frac{V_t}{W_f} \times W_t \ln \frac{W_1}{W_2} = RF \ln \frac{W_1}{W_2}$$

where W_1 = gross weight at start of cruise
and W_2 = gross weight at end of cruise

PRE-FLIGHT PREPARATION:

The optimum W/δ to be flown for this test is determined from the speed power at constant W/δ tests. Specifically, the data as plotted in Figure 3.20 is used to deter-

mine the optimum W/δ and Mach number to fly to obtain a maximum range factor and thus, maximum range.

To find these optimum values to be flown, enter Figure 3.20 and read specific range and Mach values at the peaks of the W/δ curves. From the specific range values, determine range factor using V_t, W_f, and W_s. (W_s is the standard weight at which the W/δ test was flown). Plot values of range factor vs W/δ as shown in Figure 3.21.

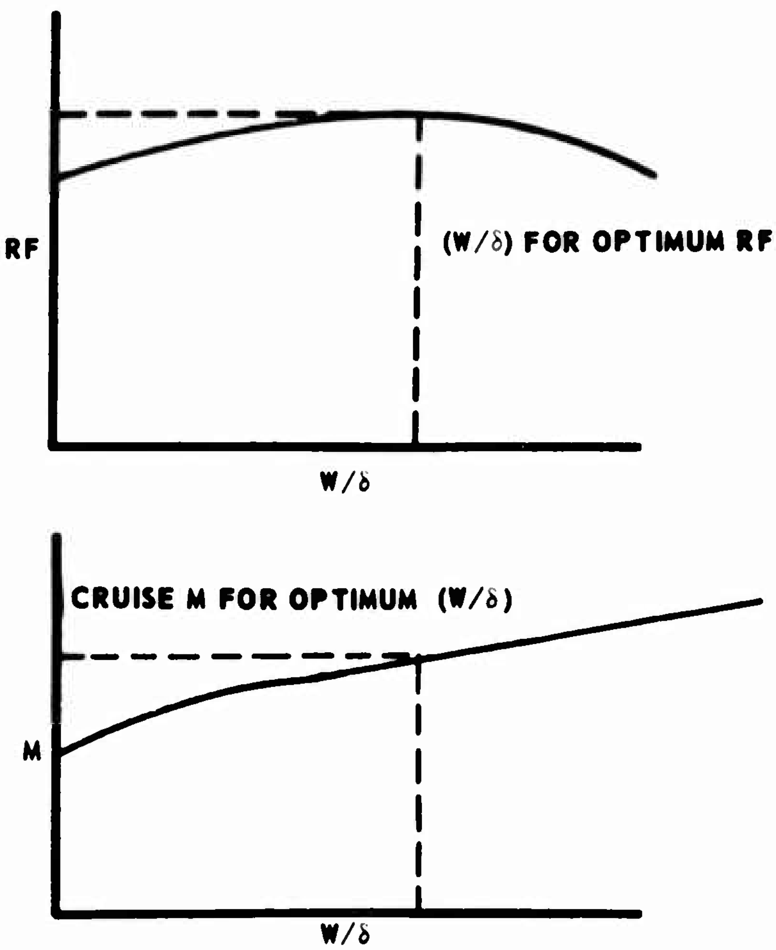


FIGURE 3.21

The peak of the RF versus W/δ curve yields the optimum W/δ to fly. The corresponding Mach number to fly can be obtained from M versus W/δ.

Usually a series of flights will be flown at a W/δ above and below the predicted optimum W/δ.

Planning of the time or distance flown during the cruise portion of the test is outlined below:

	Fuel Used	Fuel Remaining
1. Prior to Eng start		
2. Engine start + taxi		
3. Takeoff and accelerate to climb schedule		
4. Climb		
5. Cruise		
6. Fuel Reserve		

Estimated fuel used for engine start, taxi, takeoff and acceleration to climb schedule is obtained from manufacturer's charts. Fuel used in the climb is obtained from the check climb test. Aircraft characteristics and weather conditions determine the fuel reserve. The total of these fuel increments subtracted from the total fuel available gives the amount of fuel available for the cruise portion of the test. The flight route to be followed will be dictated by weather conditions and winds aloft and should be a single course out and the reciprocal back to the base.

Below is a suggested flight data card to be used by the pilot.

Pilot____________ A/C No.______ Date_______

W/δ = ____________

Data Point	Time	F/C	V_i	H_i	T_i	RPM	EGT
Prior Eng Start							
Start							
Taxi							
T.O.							
Start of climb							
End of Climb							
Start Cruise							

When on cruise schedule, record data often enough to obtain at least 10 points.

Fuel counter vs altitude chart for the desired W/δ is prepared as outlined in the pre-flight section of the speed-power at constant W/δ test.

IN-FLIGHT TECHNIQUES:

The following is a recommended procedure for performing the range cruise control test:

(1) Prior to engine start, check that the correct amount of fuel is on board and that the fuel counter is set correctly.

(2) Record data at each point planned, i.e., engine start, taxi, takeoff, start of climb and end of climb.

(3) Upon reaching the altitude that corresponds to the fuel counter reading for the desired W/δ set up the cruise climb at the desired Mach number.

(4) Increase altitude as the fuel counter is decreased to maintain a constant W/δ. An alternate method is to hold a constant altitude and step climb the aircraft in increments of 100 to 200 feet. Mach number will be held constant throughout the cruise climb. This will require a slight decrease in RPM if below the tropopause and nearly a constant RPM above the tropopause. Examination of Figure 3.22 will show that for a given W/δ and Mach number a constant $N/\sqrt{\theta}$ is required. Engine RPM is decreased as T_a decreases to hold $N/\sqrt{\theta}$ constant.

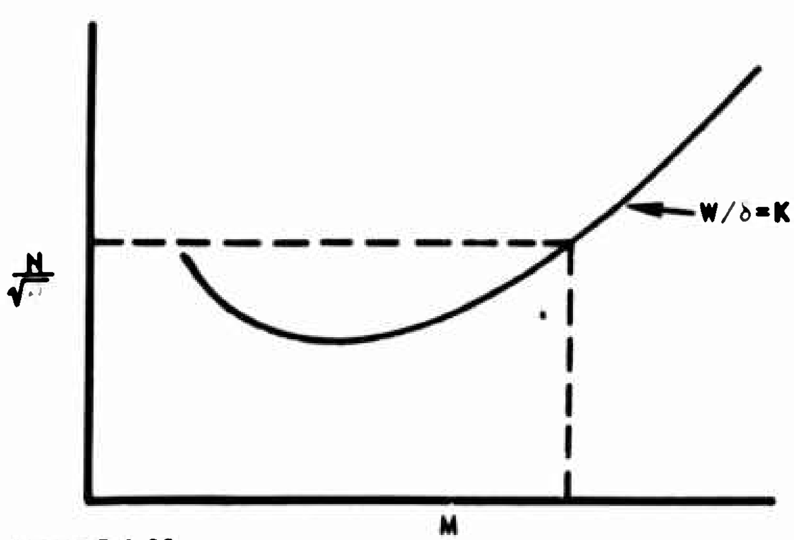


FIGURE 3.22

DATA REDUCTION OUTLINE:

Determine V_c, H_c, M, T_{ic}, N_{ic} and EGT at each data point recorded during the cruise portion of the test then proceed as described below:

Column	Symbol	Units	Description
1	W_{empty}	Pound	Basic aircraft weight includes pilot, oil, hydraulic fluid and oxygen
2	Fuel Remaining	Pound	Total fuel remaining in pounds at each data point
3	W_t	Pound	Aircraft total weight at each data point
4	Fuel used	Pound	Total fuel used
5	Time	Minutes	Total elasped time from engine start
6	ΔTime	Minutes	Elapsed time between data points
7	$V_{t_{tavg}}$	Knots	Average test true airspeed between consecutive data points
8	ΔR	N.M.	Nautical air miles traveled between consecutive data points = $\Delta T \times V_{t_{avg}}/60$
9	$\Sigma \Delta R$	N.M	Total nautical air miles traveled
10	NAM/lb		$\frac{V_t}{W_f}$

Plot H_{ic}, T_a, N_{ic}, EGT, W_t, Fuel Used, V_t, NAMT, vs Time

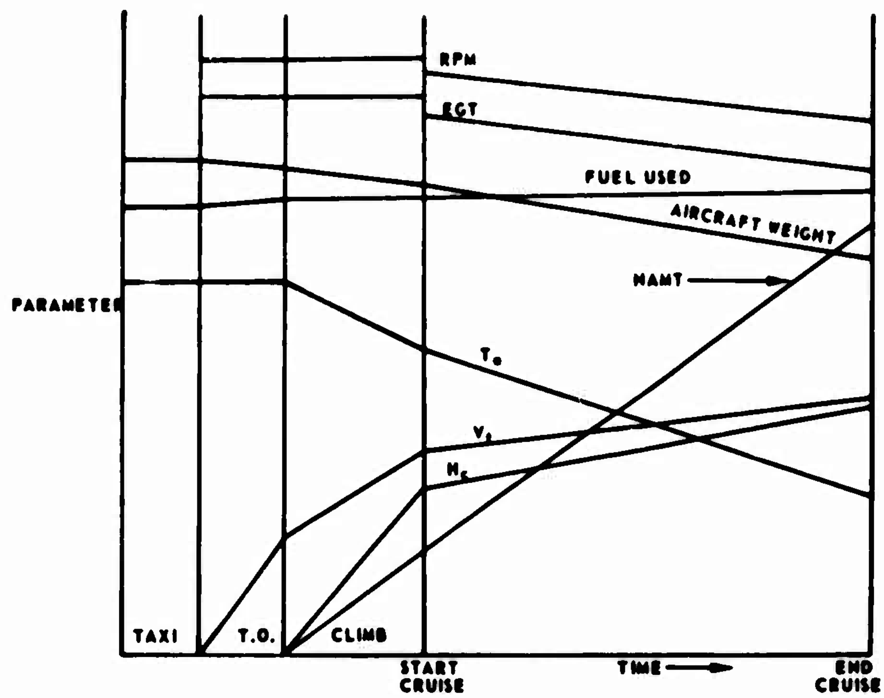


FIGURE 3.23

Step (11) is computed for each data point during the cruise portion of the mission.

11	RF	N.M	Range Factor = $\frac{V_t}{W_f} \times W$

Plot (RF) on Figure 3.21

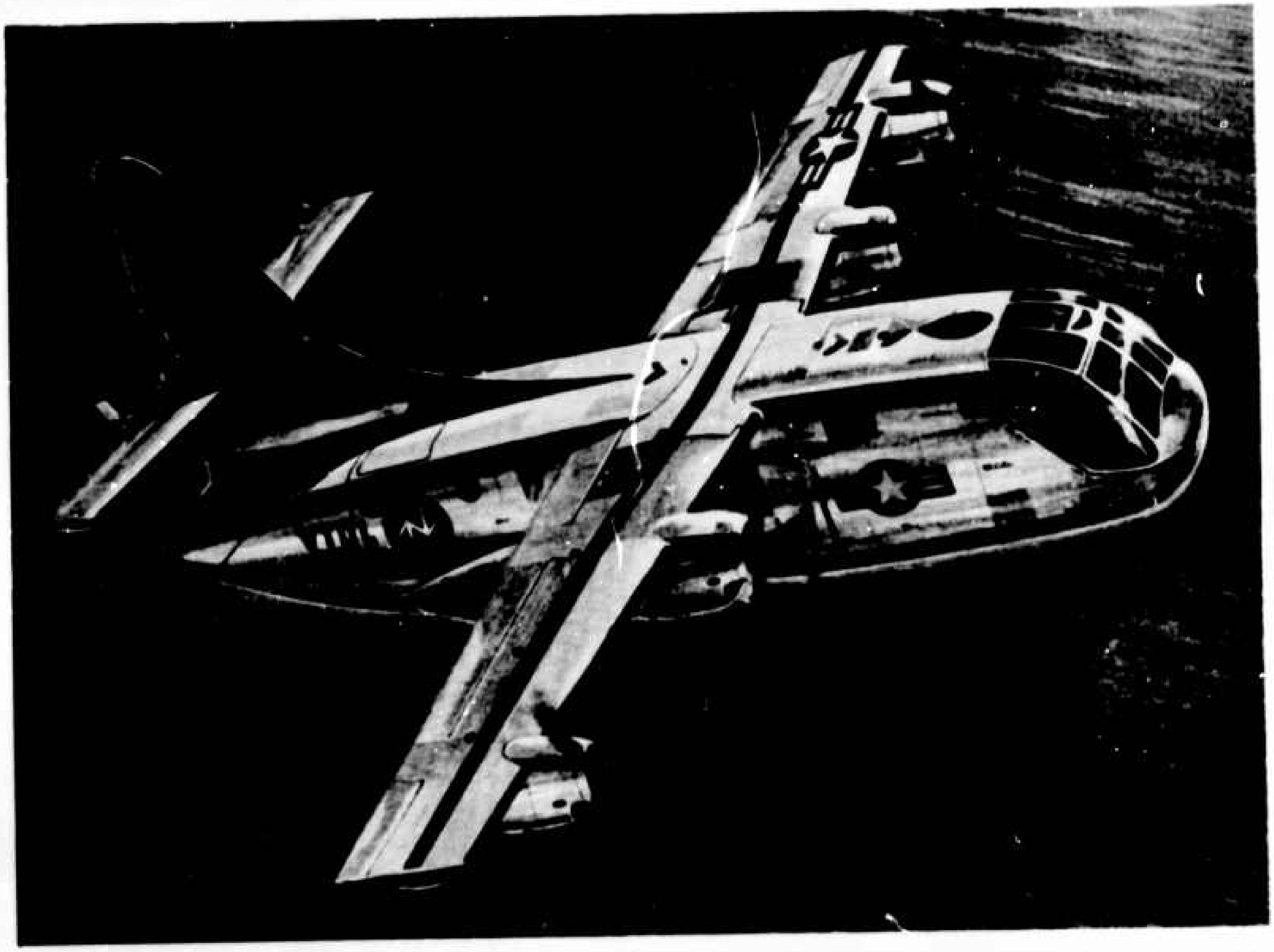
12	W_1	pound	Initial gross weight at start of cruise
13	W_2	pound	Final gross weight at end of cruise
14	$\ln \frac{W_1}{W_2}$		
15	Range Cruise	N.M	Computed distance traveled during cruise portion of the test
16	R.F.	N.M.	Range factor = $\frac{\text{Range}}{\ln \frac{W_1}{W_2}}$

Plot (16) on Figure 3.21 as a check on (11) and on standard day performance data.

The total range capability of the aircraft can now be evaluated. Total Range = NAMT during climb + NAMT during cruise.

Distance traveled during the climb is obtained from the check climb test and distance traveled during the cruise is computed by using the maximum RF and $\ln \frac{W_1}{W_2}$ where W_1 is the weight at the start of cruise and W_2 is the weight at minimum fuel reserve.

Compute the total range of the aircraft by this method and compare this value with that given in the pilot's handbook.



CHAPTER IV

TURNING PERFORMANCE

4.1 THRUST LIMITED STABILIZED TURNING PERFORMANCE

The general method of performing the stabilized turning performance test is to fly the test aircraft in a level turn at either MIL or MAX power at constant airspeed and determine the time required to turn through 360°. Hand recorded data is satisfactory although a photo panel may be employed to obtain fuel data for computer data reduction. The airspeed and altitude should be average values throughout the turn and therefore must be hand recorded in every case. Time may be recorded by any means, but hand recording is satisfactory.

DATA CARD:

A data card should be drawn up to record all necessary data. The format suggested below is satisfactory for hand recorded data.

Test Altitude__________ Configuration___

Aim V_i	V_i	H_i	Time (360°)	FAT	RPM	F/C
300						
280						
etc.						

Airspeeds should be chosen throughout the speed range of the aircraft at the altitude being flown. For a low performance jet aircraft a good speed selection would be:

1. V_{max} minus 5K
2. V_{max} minus 15K
3. -

Intervals to obtain 6 - 10 points at the test altitude

10. V_{min} plus 10K

In addition level flight points should be taken at V_{max} and V_{min} to complete the curves. All points should be flown at full power.

For a higher performance type of aircraft with afterburner it may be convenient to record time and fuel by photo panel and to fly points at each speed in MIL and MAX power. A satisfactory data card arrangement would be:

Aim V_i	Power	V_i	H_i	C.N.
350	MIL			
350	MAX			
300	MIL			
300	MAX			
etc.	etc.			

For a high performance aircraft wider intervals of aim airspeed, beginning 10 - 15K below V_{max}, are more convenient.

IN-FLIGHT TECHNIQUE:

Upon reaching test altitude a maximum speed level flight point and a minimum speed stabilized point are recorded. The aircraft is then set up for the first turn a few knots below V_{max}. The aircraft is stabilized in level flight at the desired speed, then is rolled smoothly into the turn with power advanced as load factor increases to maintain airspeed. Once stabilized in the turn, airspeed in maintained primarily by adjusting the load factor, while altitude is maintained primarily by adjusting bank. Any adjustment of either will of course interact with the other.

The external horizon must be used as the primary reference if a good stabilized turn is to be maintained. The most successful technique is to concentrate on keeping some reference point on the air-

craft nose tracking parallel to the horizon. If an adjustment in altitude is required the bank is altered to establish a new track. If an adjustment in airspeed is required the load factor can be changed by the smooth application of a change in stick pressure. In either case, corrections should be made immediately but slowly and steadily rather than abruptly. The sooner a change in airspeed or altitude is noted and corrected the smaller the correction will be. In high performance aircraft increases in airspeed must be monitored closely, as they are difficult to correct.

As in most other tests, it is far more important to maintain a constant airspeed than to be exactly on aim airspeed. A constant speed anywhere within 10 knots of the desired is satisfactory except near V_{max} or V_{min}.

The object during the turn will be to keep the following quantities constant in decreasing order of importance:

1. Airspeed
2. Load factor - with smooth changes
3. Altitude (it is important to end the turn at the same altitude at which it began).
4. Bank angle
5. Sideslip

The importance of keeping airspeed constant is based on the fact that induced drag changes with the square of the airspeed. Smooth changes in load factor are also important because of their effect on induced drag.

Once stabilized in the turn, a prominent landmark is chosen and as some reference point on the aircraft passes it, timing is begun. Timing stops when the reference passes the landmark again. The landmark should be chosen as near the horizon as possible and must be distinctive as well as clear. Lakes, green fields and lone hills are excellent. Clouds and mountain tops are good only if they are isolated.

Points should be flown covering the complete speed range at each desired power setting. However, minimum speed points in MAX power may have to be omitted as impractical in some cases.

Should the load factor of the aircraft reach the structural limit as speeds are decreased, the flight should be discontinued from this approach. Points should then be taken at near V_{min}, increasing speed from point to point until limit load factor is again reached.

■ DATA REDUCTION OUTLINE:

The outline given here is applicable to the T-33A aircraft using engine charts of $N/\sqrt{T_a}$ vs F_n/δ_{t_2} and Mach number. The procedure for other types of aircraft will be similar, but modification will be necessary depending on type of engine, type of thrust charts available, etc.

First, for each test point determine instrument corrected values of RPM and free air temperature. Determine calibrated airspeed and altitude (average values during the turn) and find Mach number and true airspeed. Find standard air temperature at the nominal test altitude. Then proceed with the following steps:

Column	Symbol	Units	Description
1	Δt	Sec.	Time to turn 360°
2	ω	deg/sec	$360/\Delta t$
3	n_t		$n_t = \sqrt{.1087\left(\frac{V_{t_t}}{\Delta t}\right)^2 + 1}$ or from charts using ω and V_{t_t}

NOTE: If $Ta_t = Ta_s$ within three degrees, then steps 4 through 11, 14 and 16 can be eliminated.

4	$N_t/\sqrt{T_{a_t}}$		
5	$N_t/\sqrt{T_{a_s}}$		

6	$(F_n/\delta_{t_2})_t$		From engine charts of F_n/δ_{t_2} vs $N/\sqrt{T_a}$
7	$(F_n/\delta_{t_2})_s$		
8	P_{t_2}/P_a		From chart of P_{t_2}/P_a vs M for different ram efficiencies
9	$(D/\delta)_t$		$\left(\frac{F_n}{\delta_{t_2}}\right)_t \times \frac{P_{t_2}}{P_a}$
10	$(D/\delta)_s$		$\left(\frac{F_n}{\delta_{t_2}}\right)_s \times \frac{P_{t_2}}{P_a}$
11	$\Delta\left(\frac{D}{\delta}\right)$		$\left(\frac{D}{\delta}\right)_s - \left(\frac{D}{\delta}\right)_t$
12	M^2		
13	S	square ft	Wing area
14	$.000675\,\Delta\left(\frac{D}{\delta}\right)$		
15	M^2 S		
16	ΔC_D		$\frac{.000675\,\Delta\left(\frac{D}{\delta}\right)}{M^2\ S}$
17	F/C	pound	Fuel counter reading of fuel remaining
18	Wt	pound	Aircraft weight
19	δ_t		Pressure ratio from charts at H_c of test
20	$\left(\frac{W}{\delta}\right)_t$		Weight pressure ratio
21	$C_{L_{t\,apparent}}$		$\frac{.000675\, n_t\,(W/\delta)_t}{M^2\ S}$
22	$C_{L_{corr.\ apparent}}$		From C_L vs C_D plot for aircraft using C_{L_t}, ΔC_D and M.

23	$W_{t_{avg}}$	pound	Average test weight
24	δ_s		δ for aim H_c from charts
25	$(W/\delta)_s$		Standard weight pressure ratio
26	n_s		$n_s = \frac{C_{L\,corr.}\ M_t^2\ S}{.000675\ (W/\delta)_s}$
27	V_{t_s}	knots	From tables at standard M, H_c, and T_a
28	R_s	N.M.	Radius of turn from chart using n_s and V_{t_s}
29	ω_s	deg/sec	From chart using V_{t_s} and n_s

Plot M vs $C_{L_{corr.}}$, R_s, ω_s and n_s

4.2 TURNING PERFORMANCE BY THE LEVEL ACCELERATION METHOD

PRE-FLIGHT PREPARATION:

Data for this test should be recorded on film from a photo panel installation, or by some other type of recording device. The following items must be recorded:

a. Airspeed
b. Altitude
c. Free air temperature
d. Time
e. Fuel counter reading

Pre-flight preparation is identical to that covered in the level flight acceleration test.

IN-FLIGHT TECHNIQUE:

The proper technique to be utilized in flying this test is covered in Section 2.3. However, in this test the acceleration must be carried to maximum speed in order to determine the point for zero excess thrust. This test may be performed in conjunction with the level flight acceleration test.

DATA REDUCTION OUTLINE:

For each test point determine V_c, H_c, T_a and V_{t_t} at the test altitude and airspeed.

Column	Symbol	Units	Description
1	V_t	ft/sec	$1.6889 \times V_{t_t}$
2	$V_t^2/2g$	feet	
3	h_e	feet	Corrected energy height = $V_t^2/2g + H_c$
4	Time	sec	Elapsed time between start of run and particular point.
5	W_t	lbs	Aircraft test weight at each point

6 Plot h_e, W_t and V_t vs time.

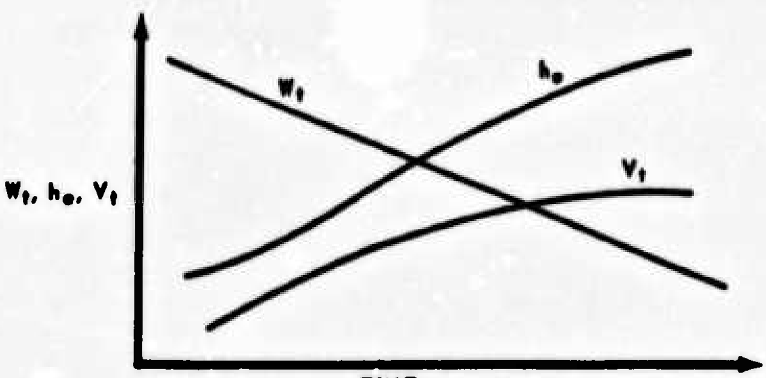


FIGURE 4.1

Column	Symbol	Units	Description
7	h_e	feet	Select arbitrary values of h_e in Figure 4.1, one value being $h_{e_{max}}$
8	dh_e/dt	ft/sec	Determine slope of curve of Figure 4.1 at each value of h_e
9	V_t	ft/sec	Airspeed corresponding to h_e from curve of Figure 4.1
10	W_t	lbs	Test weight of each value of h_e from curve of Figure 4.1
11	T_{ex}	lbs	Excess thrust = $(W/V_t) \times dh_e/dt$
12	δ		P_a/P_o at H_c
13	T_{ex}/δ	lbs	
14	M		Mach number at H_c and V_t

15 Plot T_{ex}/δ vs M for each altitude

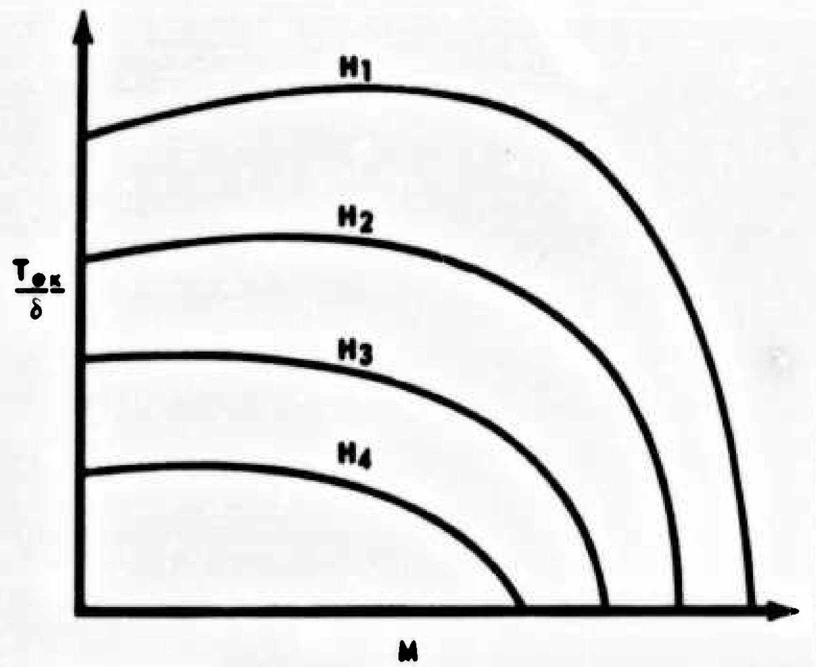


FIGURE 4.2

Column	Symbol	Units	Description
16	M		Arbitrary Mach numbers from Figure 4.2 to include all values of M at $T_{ex}/\delta = 0$
17	W_t	lbs	Total aircraft weight at (16) (From a plot of W_t vs M)
18	$nW/\delta M$	lbs	Assume n = 1
19	$(nW/\delta M)^2$	lbs^2	

20 Plot T_{ex}/δ vs $(nW/\delta M)^2$ at constant Mach numbers

NOTE: Extrapolate each constant Mach number line to determine corresponding values of M and $(nW/\delta M)^2$ at $T_{ex}/\delta = 0$

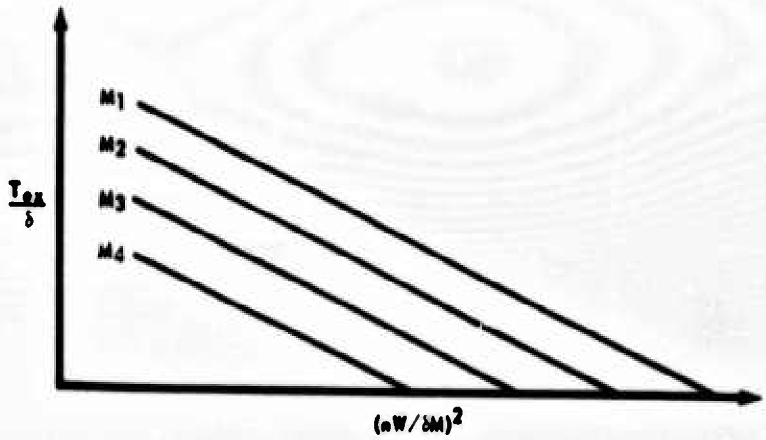


FIGURE 4.3

21	W_s	lbs	Standard weight of aircraft at desired performance altitude
22	δ		From standard tables at desired performance altitudes
23	$(nW/\delta M)^2$	lbs^2	From Figure 4.3 at $T_{ex}/\delta = 0$
24	n		Maximum load factor $= \sqrt{\left(\frac{nW^2}{\delta M}\right)} \times \frac{\delta}{W_s} \times M$
25	Plot n vs M		for all altitudes selected

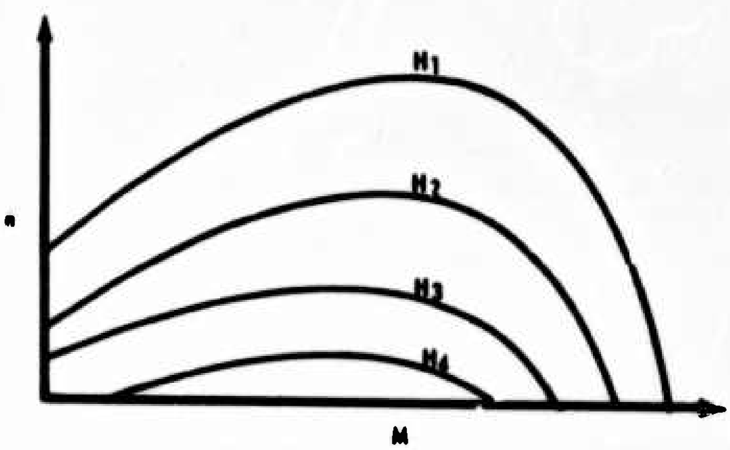


FIGURE 4.4

Column	Symbol	Units	Description
26	M		Arbitrary values of Mach number from Figure 4.4
27	V_{t_s}	ft/sec	Standard true airspeed at M and H_c selected from Figure 4.4

NOTE: Charts may be used in determining radius of turn and rate of turn.

28	V_t^2	ft^2/sec^2	
29	n		Load factor at M selected from Figure 4.4
30	n^2		
31	R	NM	Radius of turn for maximum load factor $= \dfrac{V_t^2}{6080\, g \sqrt{n^2 - 1}}$
32	Plot R vs M		

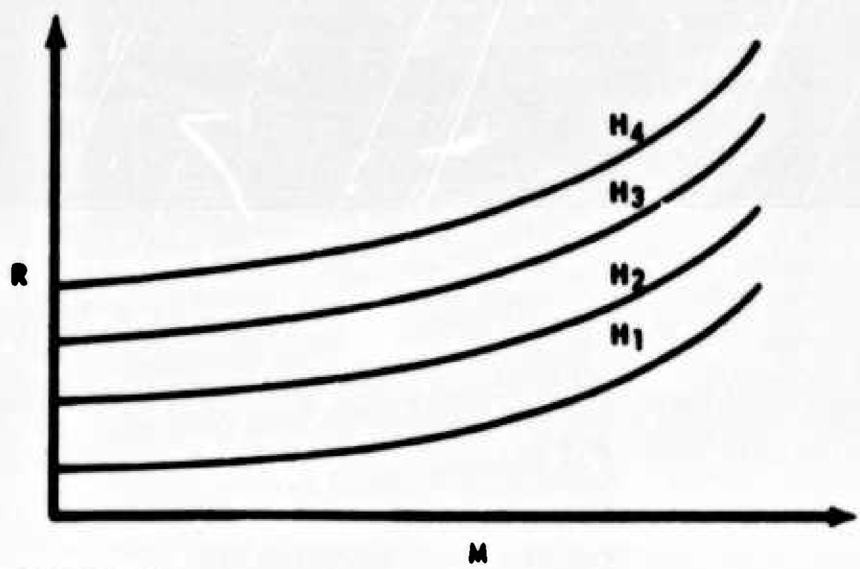
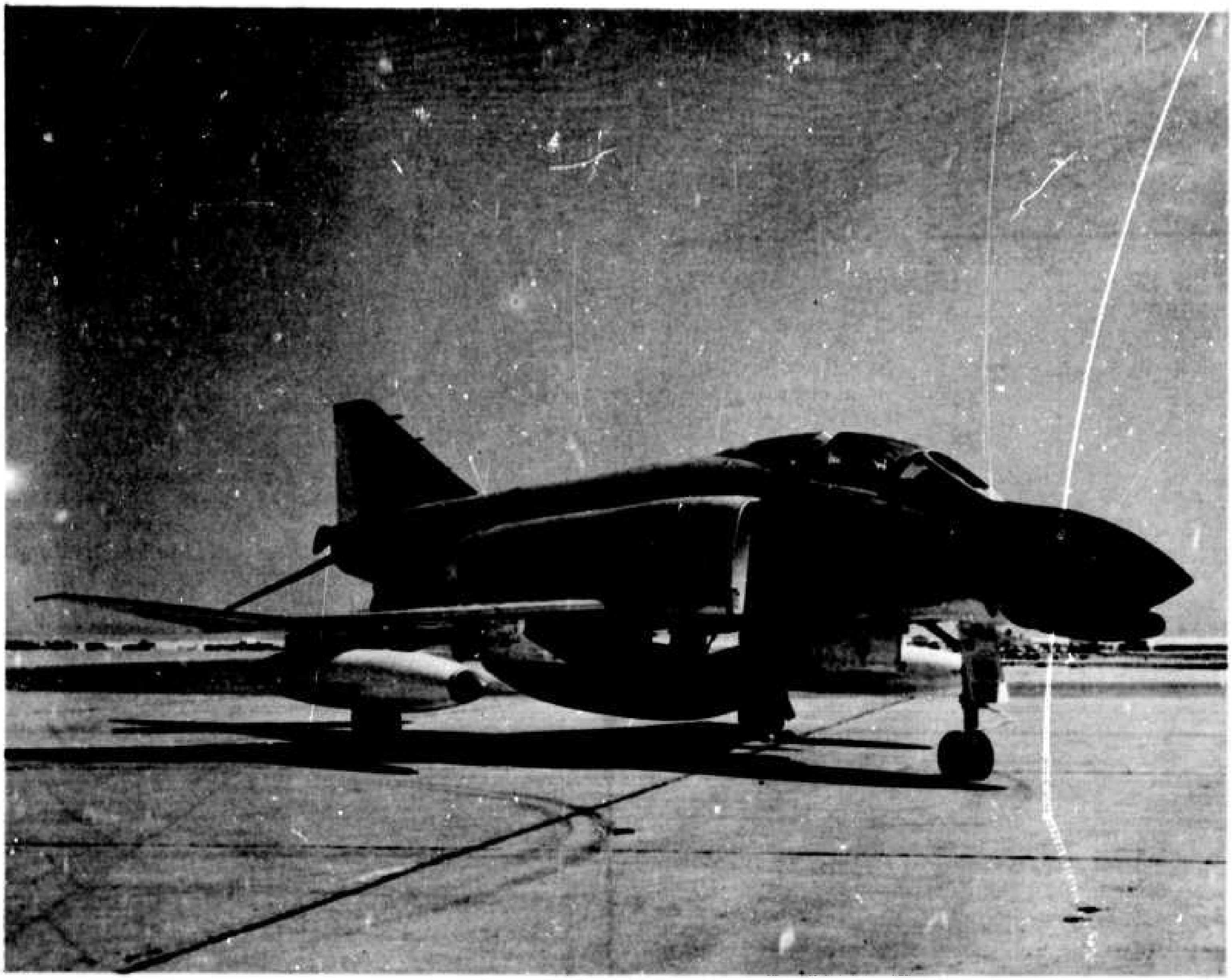


FIGURE 4.5

The remaining steps are the same as stabilized turning performance.



CHAPTER V

TAKEOFF AND LANDING PERFORMANCE TESTS

Takeoff and landing tests are important portions of the flight test program for any aircraft. During the course of a flight test program all takeoffs and landings will be recorded for data purposes whenever weather and other factors permit; and in addition a number of test missions may be devoted entirely to takeoffs in various configurations, refused takeoffs, and landings in various configurations, all done at various gross weights.

More than any other test, takeoffs and landings are affected by factors which cannot be accurately measured and properly compensated. It is only possible to estimate the capabilities of the airplane within rather broad limits, relying on a statistical average of as many takeoff and landing maneuvers as possible to cancel residual errors.

5.1 DATA RECORDING METHODS

Data recording during takeoff and landing tests is divided into two categories:

a. External Data - Ground roll, distance to 50' height, ground speed and acceleration, runway temperature, ambient pressure, and runway wind conditions.

b. Internal Data - Power parameters, V_i, H_i, T_i, EGT, etc.

The most desirable method of recording internal data is by use of a photo-recorder, however, limited hand recorded data can be taken.

External data is usually recorded by a photo-theodolite which yields distance, velocity (ground speed) and acceleration as shown in Figure 5.1.

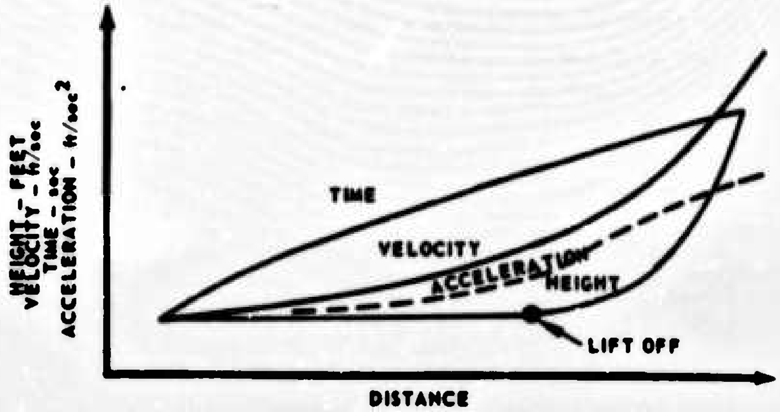


FIGURE 5.1

If a photo-theodolite is not available, the pilot or observers posted along the runway can estimate ground roll and total distance to 50' height. The main problem is being able to determine when the aircraft is passing through 50' height. In this situation a rough approximation can be made.

Another problem is determining the exact point of lift off. This point may or may not be evident from the photographic record. One of the more common methods is to observe the resulting plot of height vs distance (see Figure 5.1) and define lift off as the point where an upward trend begins. Some method of time correlating the photo-recorder and the photo-theodolite is necessary if V_{ic} and ground speed are to be used to obtain a ground effect airspeed position error correction. The usual way is to broadcast a tone that illuminates a light in the photo-panel and marks the theodolite film simultaneously. Another method is to mount a light in the side of the aircraft so that it will show on the photo-theodolite film. This light is in parallel with another light in the photo-recorder. Positive time correlation may then be established.

Temperature, ambient pressure, and

wind velocity and direction should be monitored continuously at the runway.

5.4 TAKEOFF PERFORMANCE TESTING

Since the purpose of flight testing is usually to determine the performance of the machine more than the man-machine combination, maximum performance takeoffs are usually tested quite extensively, while a more normal technique will be checked only occasionally. Maximum performance is normally achieved in a tricycle type airplane by:

1. Applying maximum power possible before releasing brakes for the takeoff roll.

2. Applying thrust augmentation at the proper point in the takeoff roll (as determined by computer studies or trial and error).

3. Leaving the nose wheel on the runway, perhaps with some back stick pressure to lighten the nose wheel load, until just a few knots before optimum lift-off speed.

4. Rotating the aircraft smoothly to the best angle of attack for lift-off and holding it there. For most aircraft it is desirable to become airborne simultaneously as the desired angle of attack is reached.

5. Maintaining the best angle of attack until 50 feet. This is done by maintaining the speed in most cases, however safety may dictate an increase in speed after lift-off.

6. Raising gear promptly when safety permits.

This procedure must be modified for various types of aircraft. Some will require that rotation be performed sooner to reduce drag. Others will have insufficient elevator power to takeoff at the optimum angle of attack and will require full back stick early in the takeoff roll.

To ensure that the optimum lift coefficient has been obtained it is desirable to measure takeoffs at a variety of speeds above and below that for $C_{L_{opt}}$. Care must be exercised when rotating at speeds below $C_{L_{opt}}$ since ground rolls can be greatly increased by this technique. Data for each takeoff must be corrected for wind, runway slope, thrust, weight, and density. (Refer to Chapter VIII of FTC-TIH 62-2005).

DATA REDUCTION:

Column	Symbol	Units	Description
1	Obtain photo-theodolite plot of height and time vs distance (See Figure 5.2)		

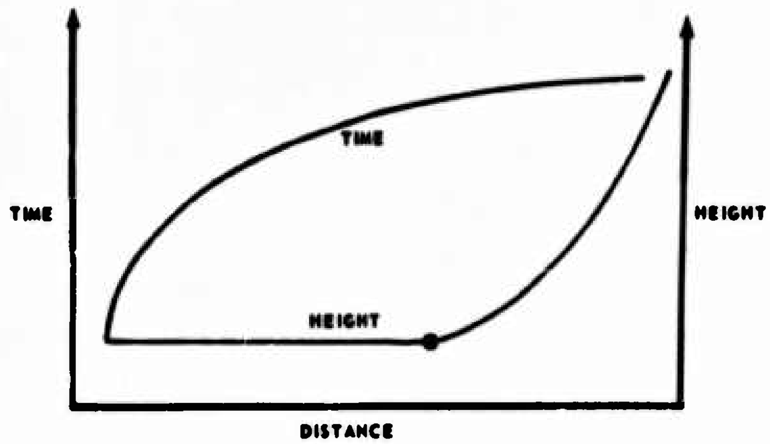


FIGURE 5.2

Column	Symbol	Units	Description
2	Wind Angle	deg	Wind direction - Runway heading
3	V_w	ft/sec	Wind velocity x cosine of wind angle
4	V_{TO}	ft/sec	Slope of time line on Figure 5.2 at moment of lift-off
5	$\left(\frac{V_{TO} + V_w}{V_{TO}}\right)^{1.85}$		
6	T_a	^oK	Observed data
7	P_a	in. Hg	Observed data
8	$\sqrt{\sigma_t}$		$\sqrt{9.625 \times P_a/T_a}$
9	$\sqrt{\theta_a}$		$\sqrt{\frac{T_a}{288}}$

10	$N_t/\sqrt{\theta_a}$	RPM	Instrument corrected RPM/ $\sqrt{\theta_a}$
11	$(F_g/\delta_a)_t$	lb	From thrust stand data at $N_t/\sqrt{\theta_a}$
12	$(F_g/\delta_a)_s$	lb	From thrust stand data at sea level standard conditions, 100% RPM
13	F_{g_t}/F_{g_s}		$\dfrac{(F_g/\delta_a)_t \times \delta_{a_t}}{(F_g/\delta_a)_s \times \delta_{a_s}}$
14	Fuel Remaining	lb	From fuel counter and fuel temperature
15	W_t	lb	Basic weight + fuel remaining
16	W_s/W_t		Standard weight from previous tests
17	$\dfrac{S_{g_s}}{S_{g_t}}$		$\left(\dfrac{W_s}{W_t}\right)^{2.3}\left(\dfrac{\sigma_t}{\sigma_s}\right)\left(\dfrac{F_{g_t}}{F_{g_s}}\right)^{1.3}$ or some other modification of ground roll equation.
18	$\dfrac{S_{a_s}}{S_{a_t}}$		$\left(\dfrac{W_s}{W_t}\right)^{2.3}\left(\dfrac{\sigma_s}{\sigma_t}\right)^{0.7}\left(\dfrac{F_{g_t}}{F_{g_s}}\right)^{1.6}$ or some other modification of the air distance equation
19	$S_{g_{t_w}}$	ft	Ground roll distance from Figure 5.2
20	S_{g_t}	ft	$S_{g_{t_w}}\left(\dfrac{V_{TO} + V_w}{V_{TO}}\right)^{1.85}$
21	Sin θ		Sine of runway slope angle.
22	S_{g_o}	ft	$\dfrac{S_{g_t}}{1 - \dfrac{64.4 \; S_{g_t}}{V_{TO}^2} \sin \theta}$
23	S_{g_s}		$S_{g_o} \times S_{g_s}/S_{g_t}$
24	$S_{a_{t_w}}$		Distance from lift-off to 50 feet from Figure 5.2
25	t		Time from lift-off to 50 feet from Figure 5.2
26	ΔS		$V_w \times t$
27	S_{a_t}		$S_{a_{t_w}} + \Delta S$ (Keep in mind if head wind or tail wind)
28	S_{a_s}		$S_{a_t} \times S_{a_s}/S_{a_t}$
29	S_{tot_s}		$S_{a_s} + S_{a_t}$
30	$V_{t_{TO}}$	Knots	$(V_{TO} + V_w)$ converted to knots

31 Plot $V_{t_{TO}}$ vs S_{g_s} for all takeoffs

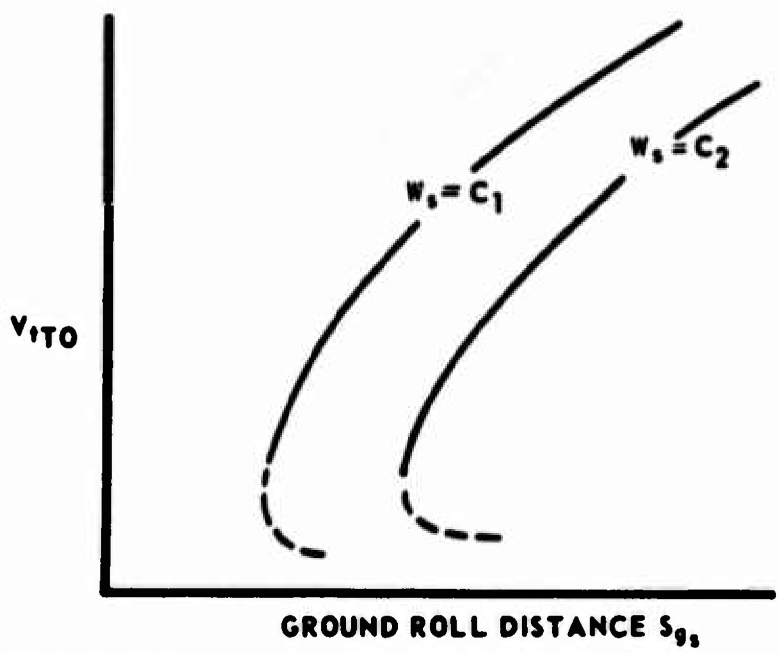


FIGURE 5.3

32	$V_{50'}$	ft/sec	From Figure 5.2
33	$V_{t_{50'}}$	knots	($V_{50'}$ + V_w) converted to knots

34 Plot $V_{t_{50'}}$ versus S_{tot_s} for all takeoffs

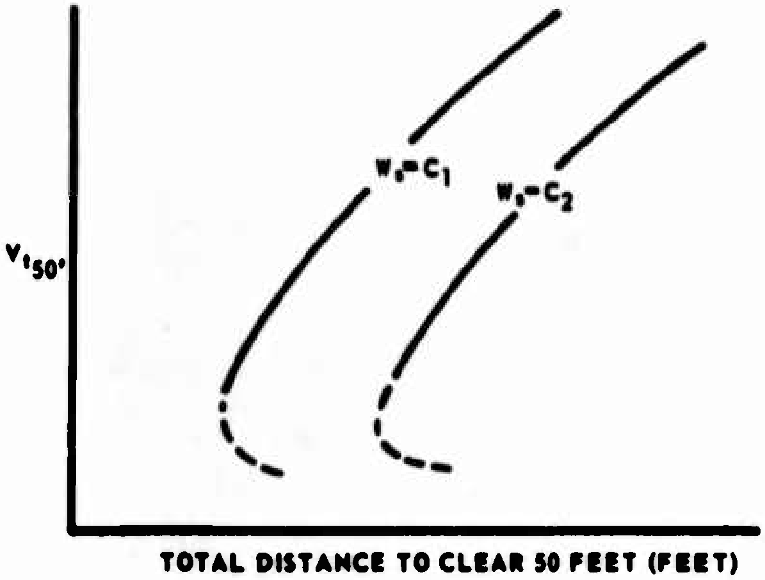


FIGURE 5.4

5.3 REFUSED TAKEOFFS

For larger aircraft which require that a great deal of energy be absorbed by wheel brakes during landing or takeoff refusal it may be desirable to perform takeoff refusals at various gross weights and from various speeds in order to determine

1. The optimum technique for minimum stopping distance

2. Runway requirements for stopping from different speeds

3. Brake energy absorption capabilities

For good data correlation it is important that the test be performed under light wind conditions (less than 5K) and of course on a dry runway. Precautions must be taken against blown tires when high levels of energy absorption are required. Correction of data to standard day conditions is usually not attempted. Comparison of test data will yield the best results.

The technique is to accelerate the aircraft on the runway to the desired speed, then chop power and apply brakes and deceleration devices in the desired order. To account for pilot reaction time a predetermined delay should be allowed after reaching the desired speed.

Most of the aims of a takeoff refusal test are achieved if distance is plotted against ground speed as recorded by a photo-theodolite. (See Figure 5.5) An event system which will mark the film when brakes are applied, drag chute is deployed, nose is lowered, etc. will be valuable. Brake energies may be calculated and plotted as well if desired, using the methods described in AFFTC-TR-61-14, August 1961.

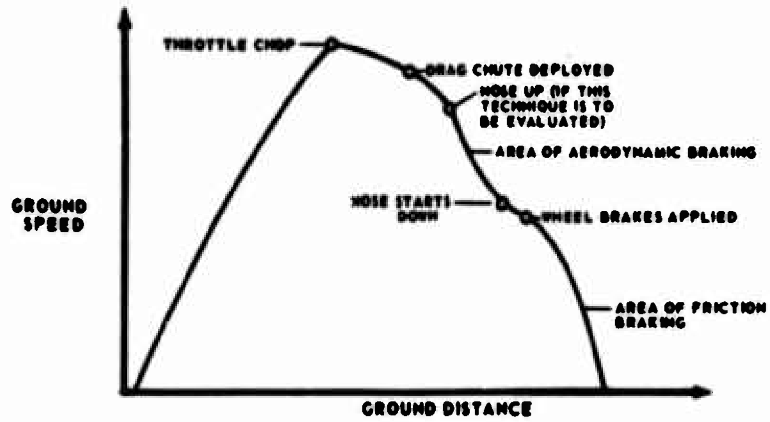


FIGURE 5.5

5.4 LANDING PERFORMANCE TESTING

The aim of the landing performance test is usually to determine the minimum stopping distance after clearing a 50' foot obstacle and after touchdown. A second aim may be equally important in early phases of testing: that of determining an optimum landing technique which is operationally feasible.

A maximum performance approach procedure will require the lowest speed and steepest angle consistent with safety. Discussion continually takes place as to what defines the minimum safe speed. In many high performance aircraft it is possible to set up a descent on the back side of the power curve with a high power setting, then flare by the addition of more power. With other types of aircraft this may not be the most desirable technique.

Touchdown technique is designed to get the aircraft on the ground solidly as soon as possible so that brakes and drag chute may be applied. A very smooth touchdown will not achieve this aim, nor will a bounce. If a high power setting was used on approach, then a power chop at touchdown will be effective, and particularly so if engine-driven boundary layer control is used.

If testing for minimum stopping distance, the procedure after touchdown will normally be to lower the nose, apply wheel brakes, extend drag devices, and apply back stick pressure just short of raising the nose to get maximum weight on the main gear. If desired, operational deviations such as raising speed brakes preparatory to barrier engagement can of course be included in the procedure. Unusual conditions such as high winds or wet runways may require a completely different procedure, but these will be examined in a separate landing test if needed.

Two variables which have a major effect on landing performance results are wind and runway surface condition. Because the test is aimed primarily at finding the capabilities of the aircraft, and because it is desired to compare results, only ideal conditions are generally considered acceptable for landing tests. Winds of 10 knots or less are required, and it is preferred that they never exceed 3 or 4 knots. A dry, uniform runway surface is generally the minimum acceptable.

5.5 DATA REDUCTION

The basic data reduction process is identical with that used for takeoff tests, using a plot similar to Figure 5.1 for determining velocity and point of touchdown. The simplifications previously mentioned should be used in the correction equations where applicable.

The correction process is complicated by the fact that results are subject to large variation with change in pilot technique. It is of great advantage if the pilot will make an attempt to be consistent in judging his approach path, flare height, rate of rotation in flare and touchdown attitude. Particular attention should be devoted to the application of brakes. The pilot should attempt to be consistent in judging light, moderate, and heavy braking force and should be meticulous about recording his observations on a data card immediately after landing.